

A REMOTELY CONTROLLED WIND TUNNEL MODEL
FOR THE DEMONSTRATION OF AIRCRAFT
STABILITY AND CONTROL CHARACTERISTICS

John Christian Merrill

NAVAL POSTGRADUATE SCHOOL

Monterey, California



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by

John Christian Merrill

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A Remotely Controlled Wind Tunnel Model
for the Demonstration of
Aircraft Stability and Control Characteristics

by

John Christian Merrill
Ensign, United States Navy
B.S.A.E., United States Naval Academy, 1974

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ABSTRACT

A remotely controlled wind tunnel model with degrees of freedom in roll, pitch, and yaw was designed and constructed to demonstrate some of the major dynamic stability and control characteristics of a full scale aircraft. The longitudinal characteristics of the model were examined, and it was found that the response to a step function input deflection of the horizontal tail could be predicted accurately.

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LIST OF SYMBOLS

a	Lift curve slope
a_t	Horizontal tail lift curve slope
b	Wing span
\bar{c}	Mean aerodynamic chord of the wing
C_{l_p}	Variation of rolling moment coefficient with roll rate perturbations
C_{l_r}	Variation of rolling moment coefficient with yaw rate perturbations
C_{l_β}	Variation of rolling moment coefficient with sideslip angle
$C_{l_{\delta_F}}$	Variation of rolling moment coefficient with vertical tail perturbation angle
C_{l_ξ}	Variation of rolling moment coefficient with aileron perturbation angle
C_{m_q}	Variation of pitching moment coefficient with pitch rate perturbations
C_{m_α}	Variation of pitching moment coefficient with angle of attack
$C_{m_{\dot{\alpha}}}$	Variation of pitching moment coefficient with time rate of change in angle of attack
$C_{m_{\alpha_t}}$	Variation of pitching moment coefficient with tail angle of attack perturbations
C_{n_p}	Variation of yawing moment coefficient with roll rate perturbations
C_{n_r}	Variation of yawing moment coefficient with yaw rate perturbations
C_{n_β}	Variation of yawing moment coefficient with sideslip angle
$C_{n_{\delta_F}}$	Variation of yawing moment coefficient with vertical tail perturbation angle

C_{n_ξ}	Variation of yawing moment coefficient with aileron perturbation angle
D	Denotes the derivative of a quantity with respect to nondimensional time
h	C.G. position, fraction of the mean aerodynamic chord
h_n	Neutral point of the model, fraction of the mean aerodynamic chord
i_A	Nondimensional moment of inertia of the model about the X axis
i_B	Nondimensional moment of inertia of the model about the Y axis
i_C	Nondimensional moment of inertia of the model about the Z axis
i_E	Nondimensional product of inertia of the model about the Y axis
l_t	Distance from the C.G. to the horizontal tail aerodynamic center
l_v	Distance from the C.G. to the vertical tail aerodynamic center
\hat{p}	Roll rate perturbation
\hat{q}	Pitch rate perturbation
\hat{r}	Yaw rate perturbation
S	Wing area
S_t	Horizontal tail area
S_v	Vertical tail area
\hat{U}	Perturbations of airspeed in the X direction
V_H	Horizontal tail volume ratio, $(l_t S_t)/(\bar{c} S)$
V_V	Vertical tail volume ratio, $(l_v S_v)/(b S)$

Greek Symbols

α	Angle of attack
α_t	Perturbation of the horizontal tail angle
β	Sideslip angle
δ_F	Perturbation of the vertical tail angle
ζ	Damping ratio
η	Perturbation of the elevator angle
μ	Friction Coefficient
ξ	Perturbation of the aileron angle
ω_n	Undamped natural frequency

Superscripts

- Variation of the quantity with respect to time

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I. INTRODUCTION

In the opinion of the author, in order for a student to realize full benefit from any classroom learning process, that process should contain, whenever possible, some type of functional visual aid or laboratory to supplement and support the in-class lecture periods. The purpose of the design and construction of the stability and control wind tunnel model described in this thesis was to provide that additional support element to the stability and control courses offered at the Naval Postgraduate School, Monterey, California.

Although it was not possible to incorporate all of the degrees of freedom of an actual aircraft into the model, the model was designed to demonstrate many of the major dynamic characteristics of a full scale aircraft. The response of the model differs somewhat from that of an actual aircraft due to the fact that it must be supported within the wind tunnel test section and prohibited from translating in any of the three coordinate functions (Fig. I-1). This difference, however, is slight and does not detract from the primary purpose of the model.

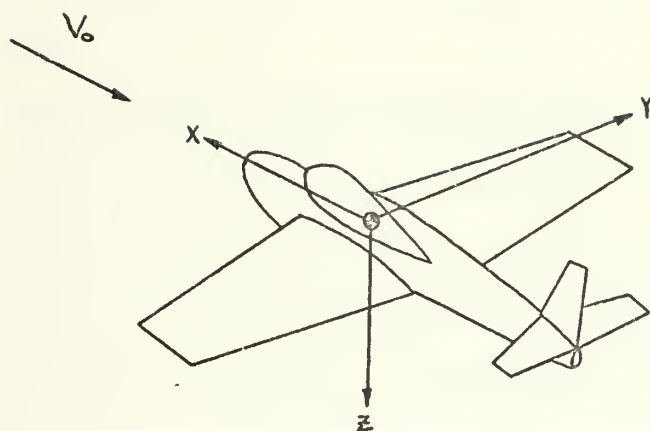


Figure I-1.

Model Reference Stability Axis System

II. MODEL DESCRIPTION

A. GENERAL FEATURES

The model (Fig. II-1) was designed to illustrate the characteristics of a typical single engine mid-wing aircraft. With minor modifications, however, the effects of various wing geometries can be studied because the model was designed with a removable wing section. Similarly, model tail sizes and shapes are variable. By the use of various wing and tail geometries the stability characteristics of the model may be changed.

Model attitude was controlled by the deflection of ailerons, and by horizontal and vertical tail deflections. The horizontal and vertical tails were designed as "all flying" surfaces so as to eliminate the need for separate elevator and rudder sections. The control surfaces are controlled from outside the tunnel by means of a remote control transmitter. A receiver mounted within the model receives the signals from the transmitter and, by means of mechanical linkage, a servo deflects the appropriate surface for control (Fig. II-2).

The resulting model motion was detected by three potentiometers which were situated so as to determine the angular rotation of the model about the X, Y, and Z stability axes. The potentiometer output was connected to a chart recorder which supplied a graphical history

of the model response. The motion sensor circuitry is shown in Figure II-3.

B. PHYSICAL MODEL DESCRIPTION

1. Fuselage

The fuselage has an overall length of 29 inches and was of circular cross section. It has a maximum outside diameter of 4-1/2 inches and a fineness rate of 6.44. The fuselage is constructed from a block of mahogany which was hollowed to give a wall thickness of 1/4 inch.

2. Wing

The wing is constructed of mahogany and has an NACA 2415 airfoil section, 6 degrees of dihedral, and no geometric twist. The wing has a span of 30 inches, a leading edge sweep of 23.5 degrees, and a taper ratio of .3. The left and right wing panels are attached to the fuselage by means of a carry through section (Fig. II-4) which also houses the main support bearing (Fig. II-5). The rectangular holes in the carry through section were cut to reduce the weight of the unit.

Ailerons, which are 8 inches long and 1 inch wide, were cut from the trailing edge of each of the wing panels. The total aileron area is 13.33 per cent of the wing area.

3. Horizontal and Vertical Tails

Both horizontal and vertical tail surfaces are constructed of mahogany and have an NACA 0012 airfoil section. The tails are designed as slab surfaces so as to

eliminate the necessity for separate elevator and rudder sections. The tail surfaces are held stationary on rotating axles by means of set screws. The axles are in turn supported by the ball bearing tail unit assembly shown in Figure II-6.

The horizontal tail has an overall span of 12.8 inches with a leading edge sweep of 25.2 degrees and a taper ratio of .24. The horizontal tail area is 26 per cent of the wing area.

The vertical tail height (from the fuselage centerline) is 6-1/2 inches, the leading edge sweep is 50 degrees, and the taper ratio is .31. A vertical tail area to wing area ratio of .22 is used.

C. MAIN SUPPORT BEARING

The model is supported by the gimbaled arrangement shown in Figure II-5. This unit is installed so that the pivot point of the bearing is located exactly 14-1/2 inches aft of the nose. This point serves as the origin for the model reference axes as well as the model center of gravity. Potentiometers incorporated within this unit detect model development in roll and pitch up to angles of 25 degrees. These potentiometers are used as voltage dividers, and their output voltage is connected to a chart recorder which provides a graphical measurement of the model response.

D. TAIL UNIT ASSEMBLY

The assembly shown in Figure II-6 serves to support the horizontal and vertical tail shafts which rotate the surfaces to the desired deflection angle. The unit is constructed of aluminum and contains four ball bearings (2 horizontal and 2 vertical) on which the tail axles are supported. The entire assembly is attached to the fuselage at a point such that the horizontal axle is exactly 12 inches aft of the model center of gravity. This resulted in a horizontal tail volume ratio, V_H , of .732, and a vertical tail volume ratio, V_V , of .08.

E. WIND TUNNEL MOUNT

The model was supported in the wind tunnel test section by means of the mounting assembly shown in Figure II-7. This support was constructed of aluminum and was attached to the roof of the wind tunnel as shown in Figure II-8. It has an overall length of 12 inches of which approximately six inches protruded down into the tunnel flow. This portion of the support was shrouded by a symmetrical aerodynamic fairing.

The model was connected to the mounting unit through a 7/16 inch diameter stainless steel tube. The shaft of the main support bearing was inserted into the bottom of the tube and tightened in place with two set screws. Collars, in conjunction with thrust bearings located within the main unit, absorbed any lift, drag, and side forces of the

model while still allowing it to rotate freely about all the coordinate axes.

The shaft of a potentiometer located on top of the mounting unit was inserted into the top of the stainless steel tube and was held tight with a set screw. This potentiometer sensed yaw angle of the model, and a yaw stopper attached to the steel tube limited yaw to 30 degrees.

F. SUMMARY OF MODEL DIMENSIONS

Wing:

Area (including portion covered by fuselage)	120 in ²
Span	30 in
Chord-root	6.15 in.
-tip	1.85 in
-MAC	4.38 in
Aspect Ratio	7.5
Airfoil Section	NACA 2415
Leading Edge Sweep	23.5°
Dihedral Angle	6°

Horizontal Tail:

Area (including portion covered by fuselage)	31.1 in ²
Span	12.8 in
Chord-root	3.94 in
-tip	.93 in
Aspect Ratio	5.27
Airfoil Section	NACA 0012

Leading Edge Sweep	25.2°
Incidence (variable)	10°
Tail Volume Ratio	.732

Vertical Tail:

Area (including portion covered by fuselage)	26 in ²
Height (from fuselage centerline)	6.5 in
Chord-root	6.5 in
-tip	1.5 in
Airfoil Section	NACA 0012
Leading Edge Sweep	50°
Tail Volume Ratio	.08

Fuselage:

Total Length	29 in
Maximum Cross Sectional Area	16 in ²
Fineness Ratio	6.44

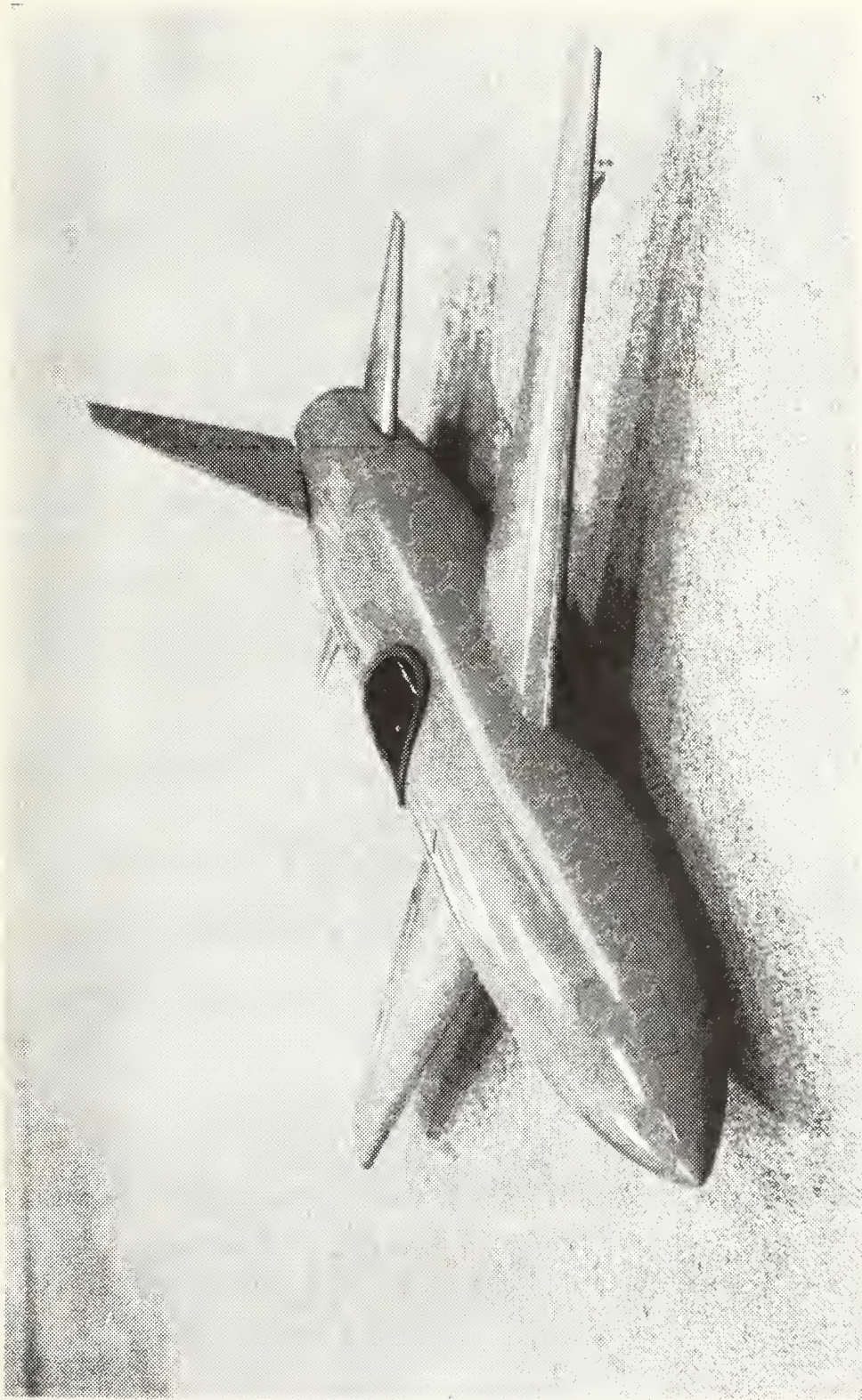


Figure II-1. Model Configuration

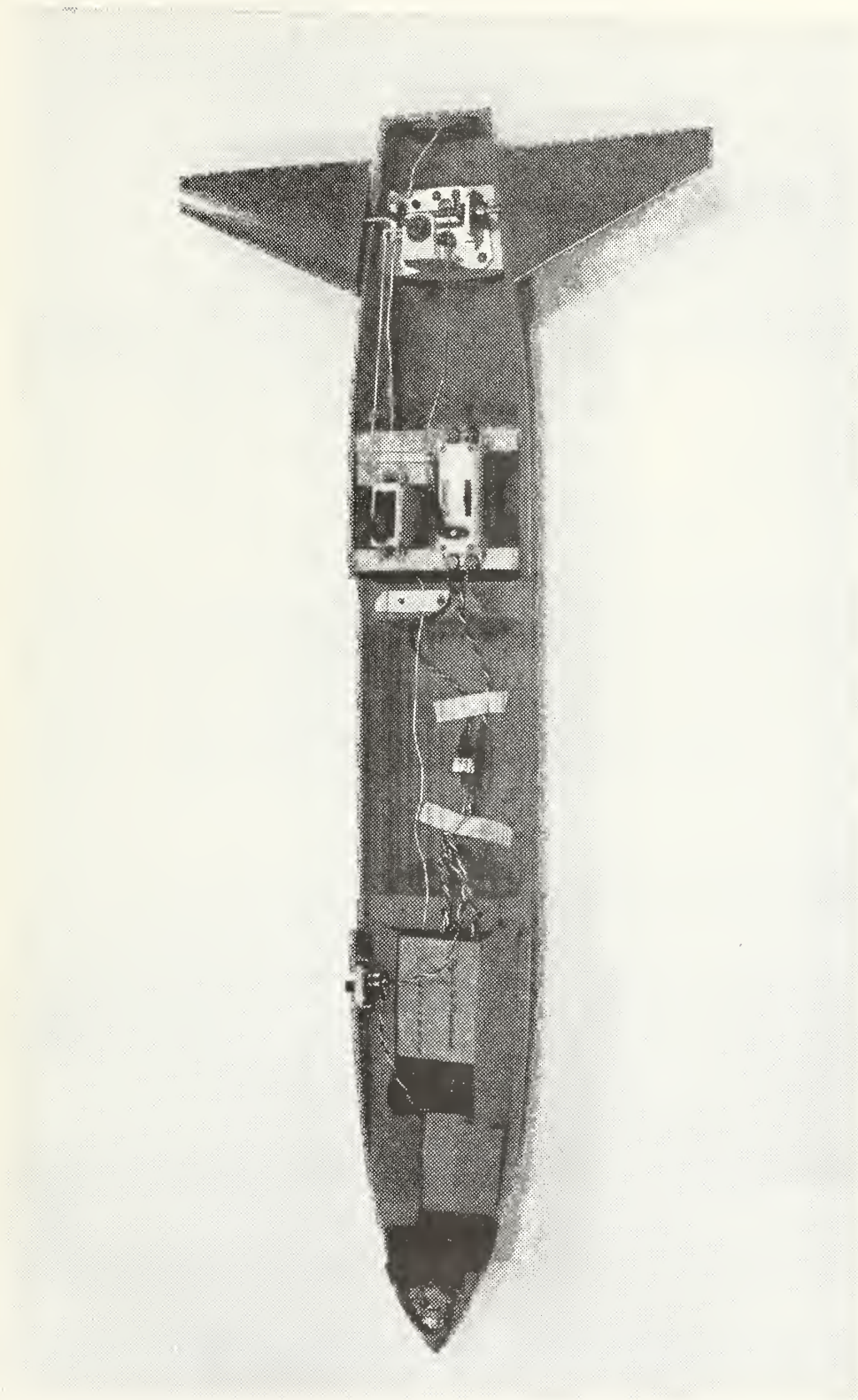


Figure II-2. Model Internal Components Showing the Receiver and Horizontal and Vertical Tail Servos

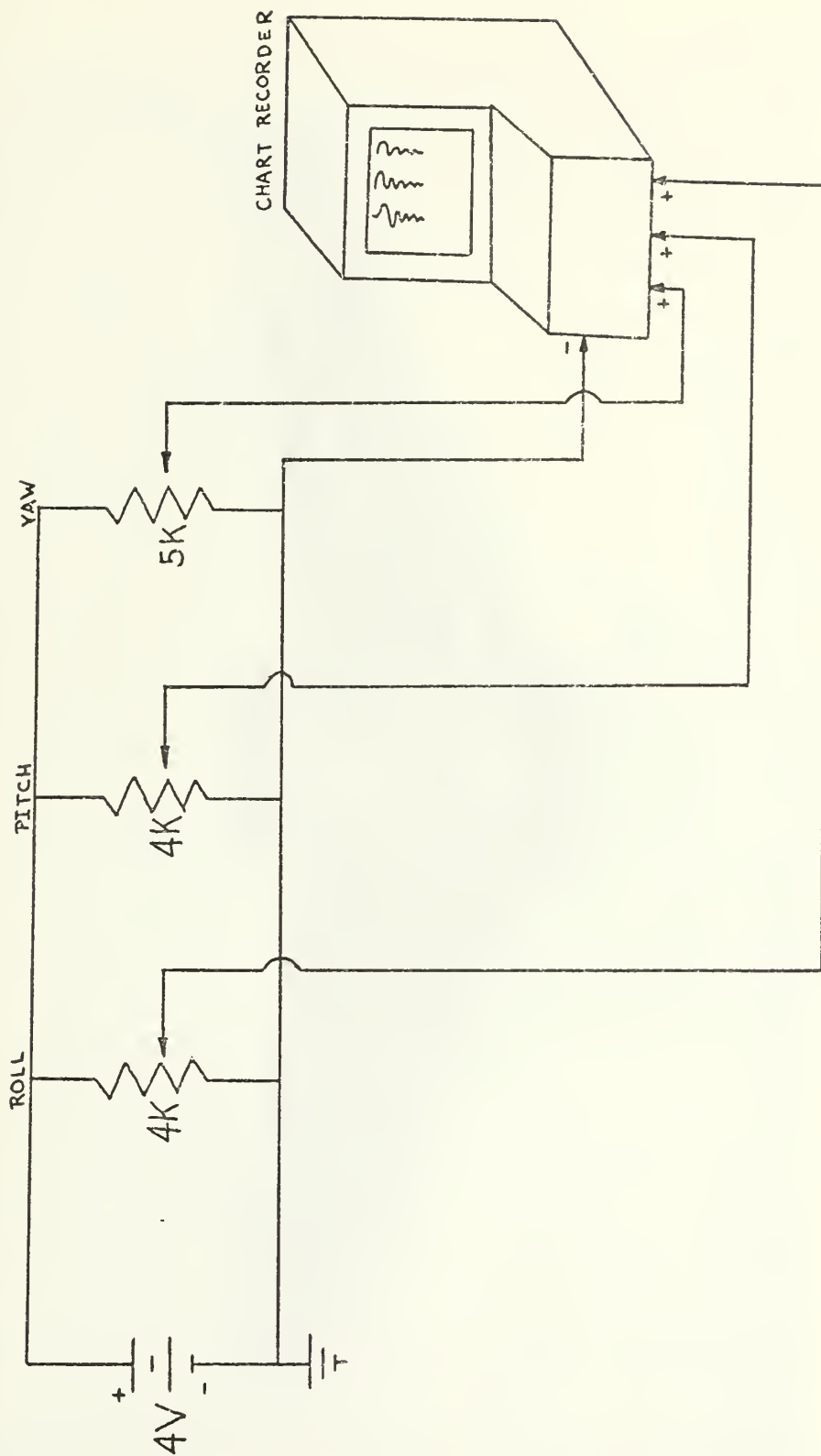


Figure II-3. Motion Sensor Circuitry

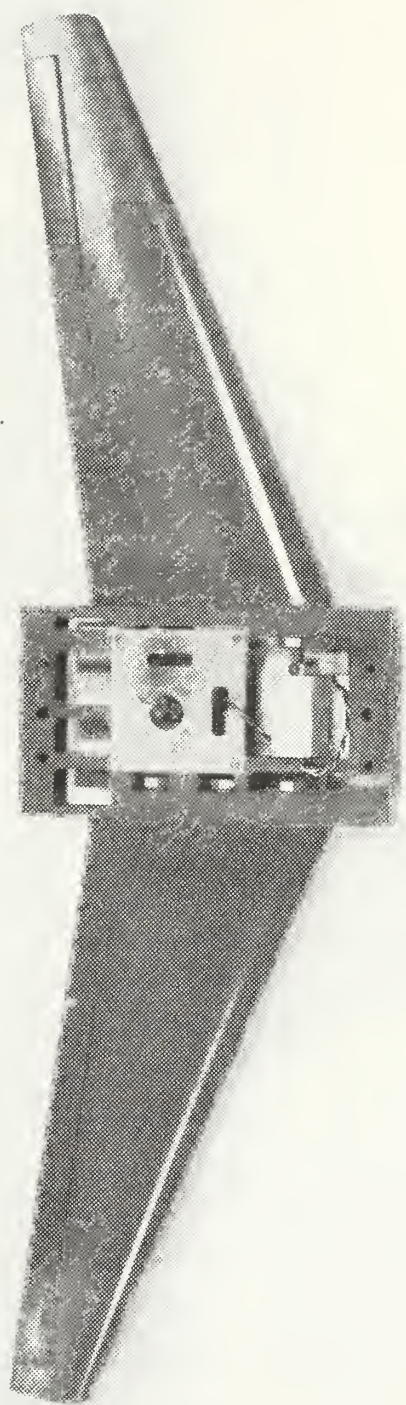


Figure II-4. Wing Section

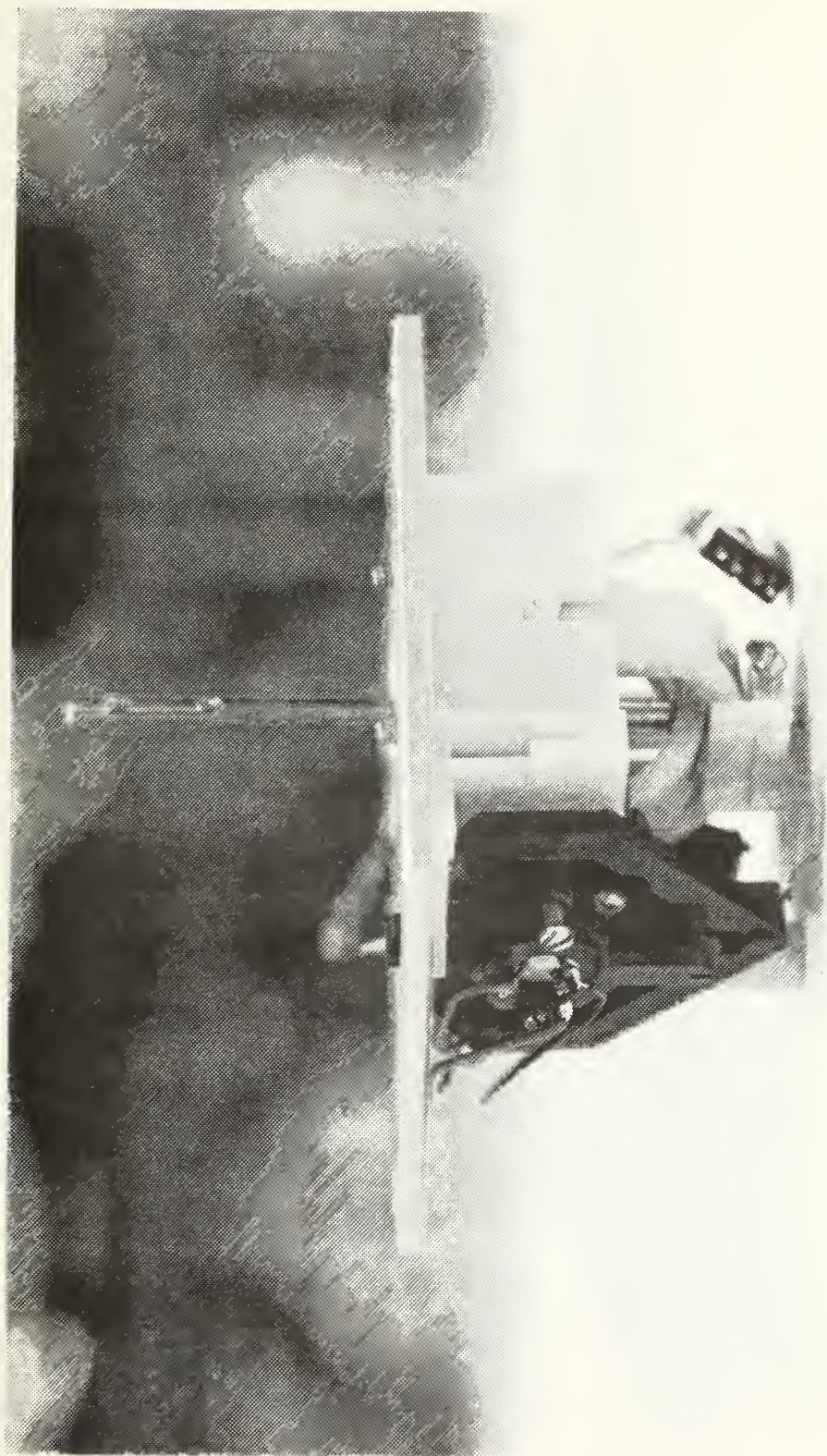


Figure II-5. Main Support Bearing

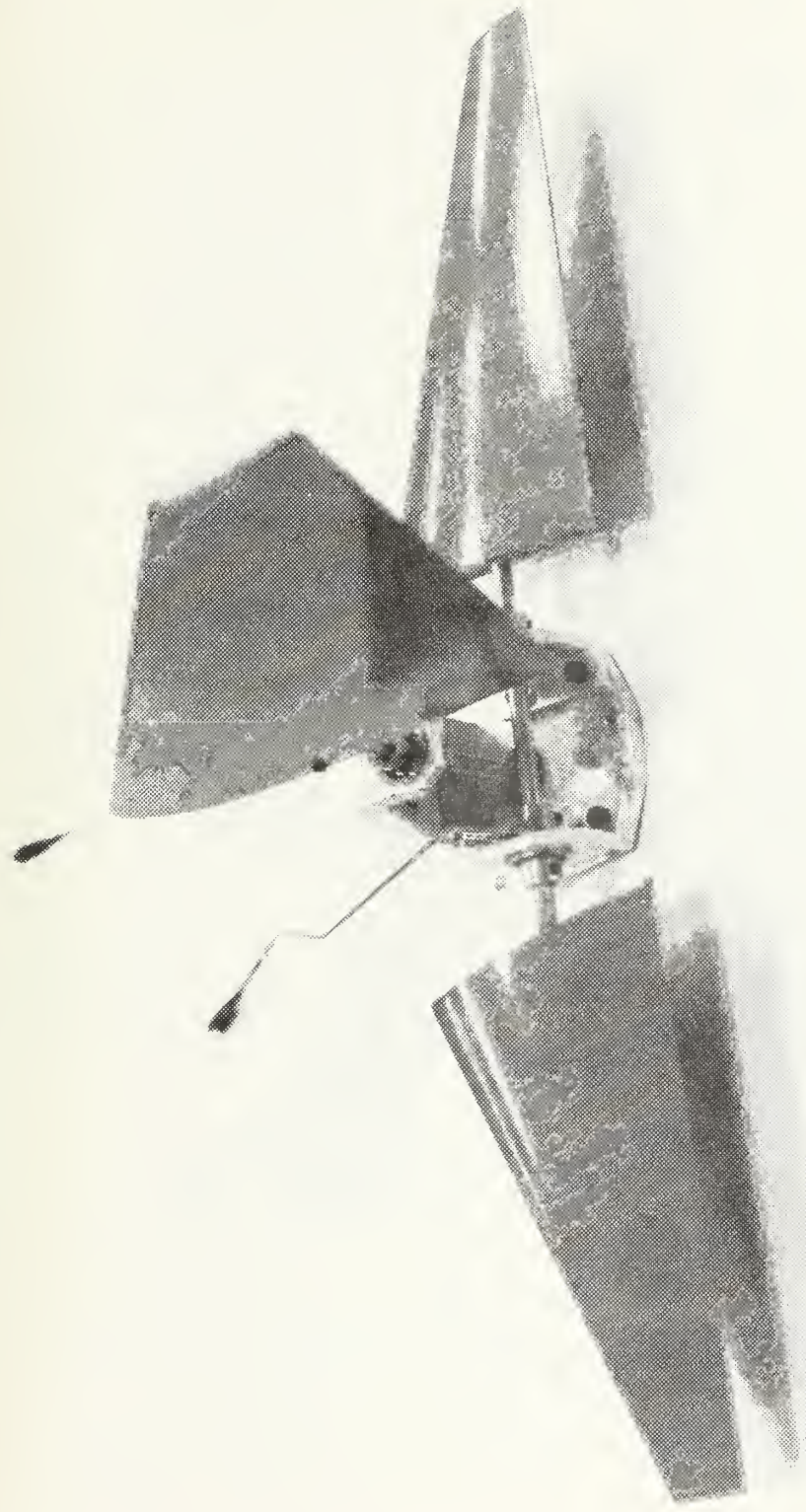


Figure II-6. Tail Unit Assembly with Tail Surfaces Installed

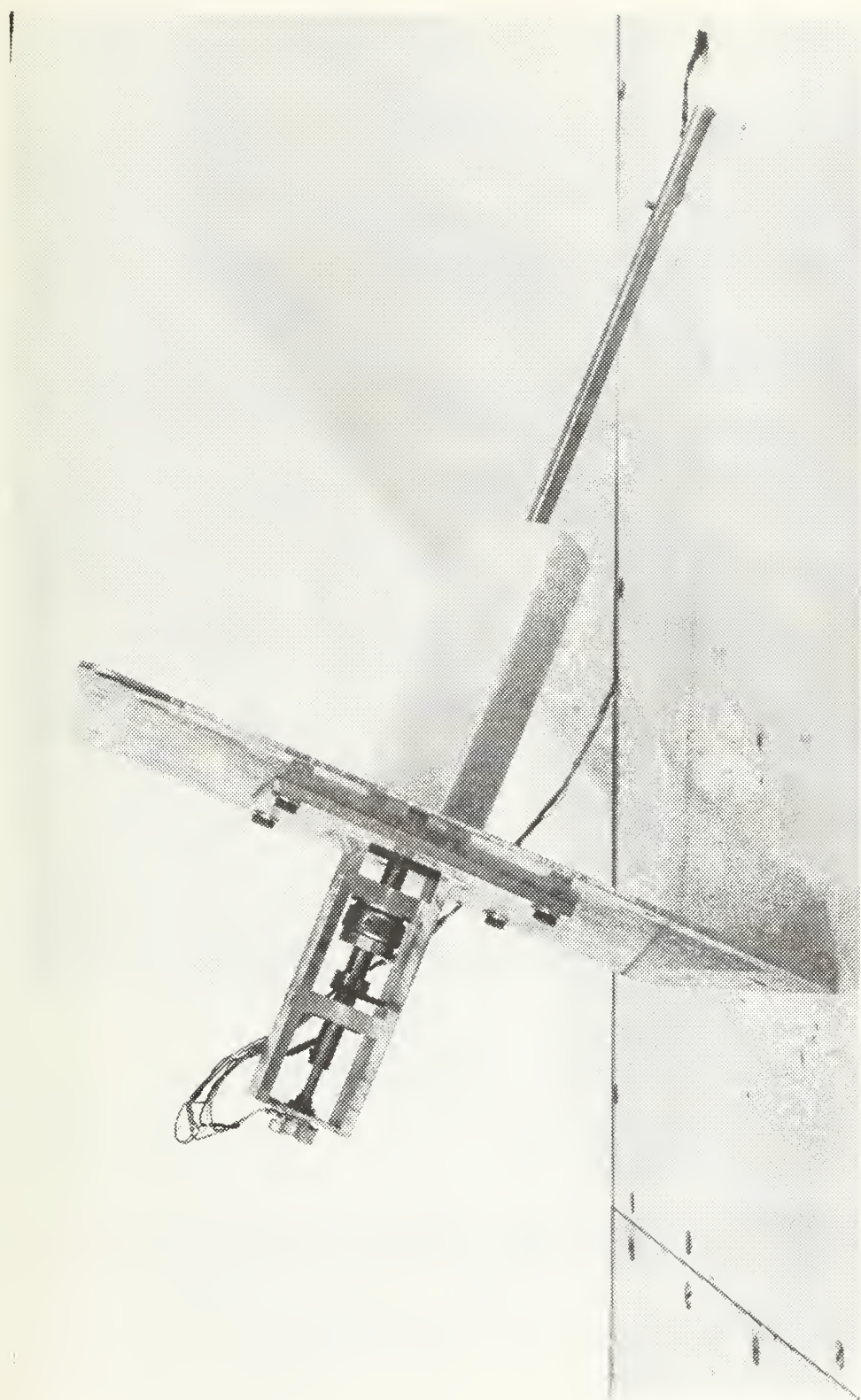


Figure II-7. Wind Tunnel Mount

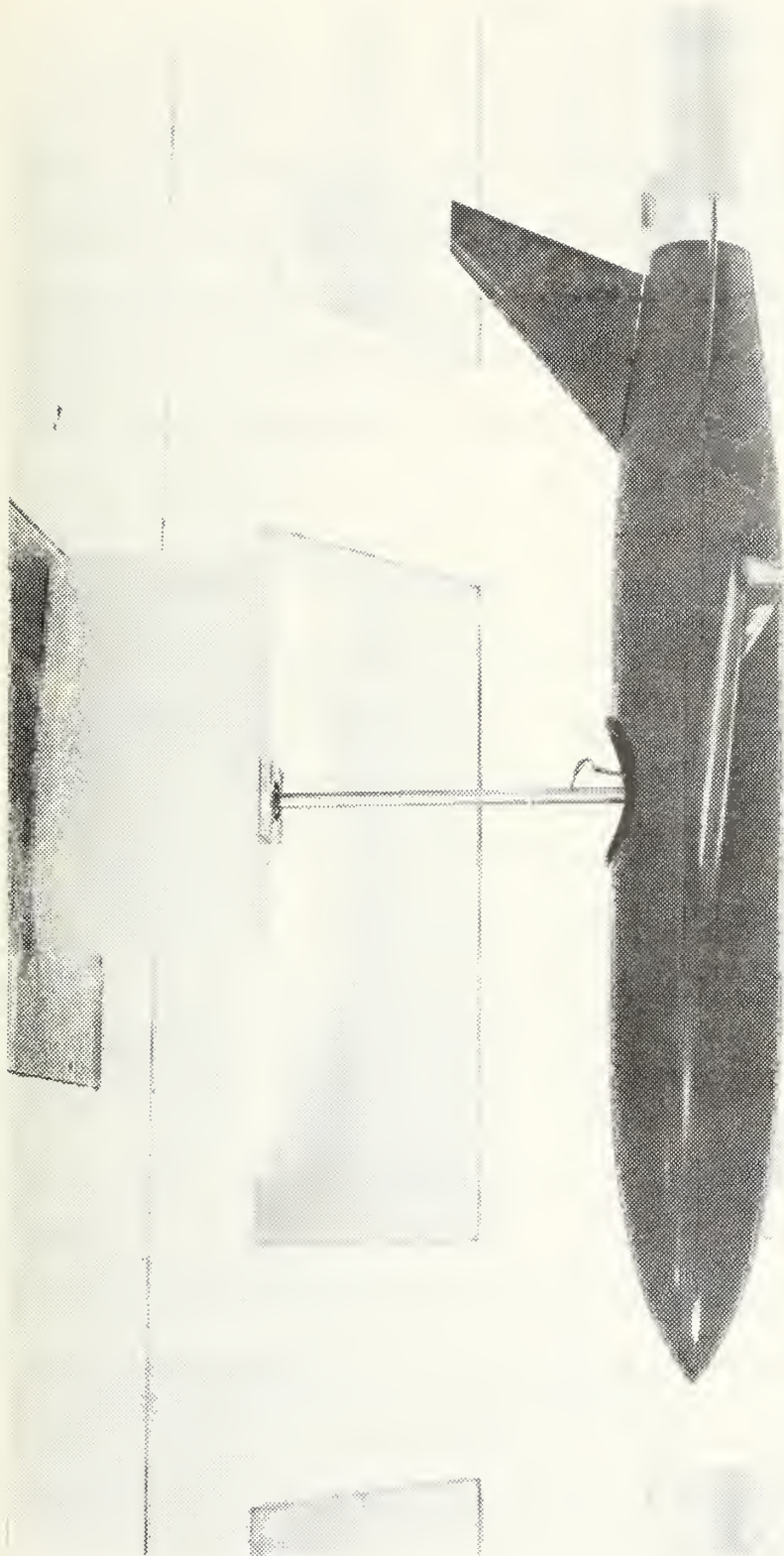


Figure II-8. Model Mounted in the Wind Tunnel

III. ANALYTICAL MODEL

The model was designed, and its motion predicted by means of linear equations of motion. From these equations it was possible to predict model response to various control surface inputs. Also, once these equations had been established, it was possible to theoretically vary model characteristics and obtain information on how this new configuration would respond.

A. LONGITUDINAL EQUATIONS OF MOTION

The nondimensional longitudinal equations of motion were obtained from Ref. 1. These equations were developed using the small disturbance theory for an aircraft having four degrees of longitudinal freedom. The model, however, was suspended in the wind tunnel so that translations along the X and Z axes were denied, thereby allowing the X and Z force equations to be neglected entirely. The elevator hinge moment equation was also neglected, as control displacement rather than control force was considered as an input.

Due to the assumptions of small angles and constant airstream velocity in the tunnel, perturbations of the velocity in the X direction, \hat{U} terms, were zero. Also, the model was situated within the tunnel so that the flight path angle, θ , and the angle of attack, α , were equal.

Due to the fact that the horizontal tail was a slab surface and there was no separate elevator section, the quantity η which was originally defined as elevator perturbation angle was replaced by tail angle of attack α_t . It was further assumed that $\dot{\alpha}_t$ terms were negligible.

Applying the above assumptions to the initial equations of motion yielded the following longitudinal equations of motion for the model.

$$i_B \ddot{\alpha} - (C_{m\dot{\alpha}} + C_{m\dot{q}}) \dot{\alpha} - C_{m\alpha} \alpha - C_{m\alpha_t} \alpha_t = 0 \quad \text{III-1}$$

$$q = \dot{\alpha} \quad \text{III-2}$$

These equations, however, do not take into account the friction inherent in the main support bearing on which the model pivots. Therefore, they must be further modified to account for this frictional retarding moment.

The frictional moment present at the pivot is a function of model supporting force. More accurately:

$$M_{\text{frict.}} = \mu_1 (W - L) \quad \text{III-3}$$

where: μ_1 = coefficient of friction (in.)

W = model weight (lbs.)

L = model lift (lbs.)

Putting this equation in nondimensional format results in the following expression:

$$C_{m_f} = \frac{2 \mu_1 W}{\rho V^2 S \bar{c}} - \frac{\mu_1}{\bar{c}} C_{L_\alpha} \alpha \quad \text{III-4}$$

where: ρ = air density (slugs/ft³)

V = velocity (ft/sec)

The first term on the right hand side of equation III-4 is the constant break-out moment of the bearing for any given airspeed; therefore, the perturbation of the frictional moment coefficient during motion was given by:

$$\Delta C_{m_f} = \frac{-\mu_1}{\bar{c}} C_{L_\alpha} \alpha \quad \text{III-5}$$

and the revised model longitudinal equations of motion are:

$$i_B \ddot{\alpha} - (C_{m_\alpha} + C_{m_q}) \dot{\alpha} - (C_{m_\alpha} - \frac{\mu_1}{\bar{c}} C_{L_\alpha}) \alpha - C_{m_{\alpha_t}} \alpha_t = 0 \quad \text{III-6}$$

$$q = \dot{\alpha} \quad \text{III-7}$$

The basic rules for the formulation of state equations from ordinary differential equations [Ref. 2] were followed in order to obtain the state variable representation of the model longitudinal equations of motion:

$$\begin{bmatrix} \dot{q} \\ \dot{\alpha} \end{bmatrix} = \begin{bmatrix} (C_{m_q} + C_{m_{\dot{\alpha}}})/i_B & (C_{m_{\alpha}} - \frac{\mu_1}{\bar{c}} C_{L_{\alpha}})/i_B \\ 1 & 0 \end{bmatrix} \begin{bmatrix} q \\ \alpha \end{bmatrix}$$

$$+ \begin{bmatrix} (C_{m_{\alpha t}})/i_B \\ 0 \end{bmatrix} \begin{bmatrix} \alpha_t \end{bmatrix}$$

III-8

B. LATERAL EQUATIONS OF MOTION

As with the longitudinal case, the lateral nondimensional equations of motion were obtained from Ref. 1 and then modified to fit the explicit conditions of the model suspended within the wind tunnel. To begin with, the Y force equation was disregarded due to the fact that the model was restricted from translating along that axis. Also, the rudder and aileron hinge moment equations were neglected since control displacement rather than control force constituted the input. The term ζ in the original set of equations denoted rudder angle perturbation; however, this was replaced by δ_F since the vertical fin was designed and constructed as a slab surface. It was further assumed that all $\dot{\xi}$ and $\dot{\delta}_F$ terms were zero.

The model was suspended within the tunnel so that the rate of change of the bank angle, $\dot{\phi}$, and roll rate, \hat{p} , were

equal. Also, the rate of change of the sideslip angle, $\dot{\beta}$, was the negative of the yaw rate, \hat{r} .

Applying the above conditions to the initial equations of motion resulted in the following model lateral equations of motion.

$$-C_{1\beta} \beta + (i_A^D - C_{1p}) \hat{p} - (i_E^D + C_{1r}) \hat{r} - C_{1\xi} \xi - C_{1\delta_F} \delta_F = 0 \quad \text{III-9}$$

$$-C_{n\beta} \beta - (i_E^D + C_{np}) \hat{p} + (i_C^D - C_{nr}) \hat{r} - C_{n\xi} \xi - C_{n\delta_F} \delta_F = 0 \quad \text{III-10}$$

$$\dot{\phi} = \hat{p} \quad \text{III-11}$$

$$\dot{\beta} = -\hat{r} \quad \text{III-12}$$

However, as before, these equations do not reflect the frictional moments present and must be modified to do so. Model side force is the major parameter contributing to the retarding frictional moments during model lateral motion. Therefore:

$$L_{\text{frict.}} = \mu_1 Y \quad \text{III-13}$$

and

$$N_{\text{frict.}} = \mu_2 Y \quad \text{III-14}$$

where: $L_{\text{frict.}}$ = frictional component of rolling moment (in-lbs.)

$N_{\text{frict.}}$ = frictional component of yawing moment (in-lbs.)

$\mu_{1,2}$ = coefficients of friction (in.)

Y = side force (lbs.)

The frictional coefficients μ_1 and μ_2 should be equal, but the coefficient for the yaw axis has a different value due to the contribution of the yaw potentiometer.

Putting these equations in nondimensional format results in the following expressions for the respective frictional moment perturbations.

$$\Delta C_{l_f} = \frac{\mu_1}{b} C_{y_\beta} \beta \quad \text{III-15}$$

and

$$\Delta C_{n_f} = \frac{\mu_2}{b} C_{y_\beta} \beta \quad \text{III-16}$$

where: b = wing span

Incorporating these contributions to the rolling and yawing moments into the previous set of equations results in the following corrected set of model lateral equations of motion.

$$\begin{aligned} & - (C_{l_\beta} + \frac{\mu_1}{b} C_{y_\beta}) \beta + (i_A^D - C_{l_p}) \hat{p} - (i_E^D + C_{l_r}) \hat{r} \\ & - C_{l_\xi} \xi - C_{n_{\delta_F}} \delta_F = 0 \end{aligned} \quad \text{III-17}$$

$$- (C_{n_\beta} + \frac{\mu_2}{b} C_{y_\beta}) \beta - (i_E^D + C_{n_p}) \hat{p} + (i_C^D - C_{n_r}) \hat{r}$$

$$- C_{n_\xi} \xi - C_{n_{\delta_F}} \delta_F = 0 \quad \text{III-18}$$

$$\dot{\phi} = \hat{p} \quad \text{III-19}$$

$$\dot{\beta} = -\hat{r} \quad \text{III-20}$$

Putting these equations in standard state variable format produced the following matrix system.

$$\begin{bmatrix} \dot{p} \\ \dot{r} \\ \dot{\beta} \\ \dot{\phi} \end{bmatrix} = \begin{bmatrix} (i_C C_{1p} + i_E C_{np})/J & (i_C C_{1r} + i_E C_{nr})/J & (i_C K + i_E H)/J & 0 \\ (i_A C_{np} + i_E C_{1p})/J & (i_A C_{nr} + i_E C_{1r})/J & (i_A H + i_E K)/J & 0 \\ 0 & -1 & 0 & 0 \\ 1 & 0 & 0 & 0 \end{bmatrix} \begin{bmatrix} p \\ r \\ \beta \\ \phi \end{bmatrix}$$

$$+ \begin{bmatrix} (i_C C_{1\xi} + i_E C_{n\xi})/J & (i_C C_{1\delta_F} + i_E C_{n\delta_F})/J \\ (i_A C_{n\xi} + i_E C_{1\xi})/J & (i_A C_{n\delta_F} + i_E C_{1\delta_F})/J \\ 0 & 0 \\ 0 & 0 \end{bmatrix} \begin{bmatrix} \xi \\ \delta_F \end{bmatrix}$$

III-21

where: $J = i_A i_C - i_E^2$

$$K = (C_{1\beta} + \frac{\mu_1}{b} C_{y\beta})$$

$$h = (C_{n\beta} + \frac{\mu_2}{b} C_{y\beta})$$

IV. STABILITY DERIVATIVE RELATIONSHIPS

In order for the equations of motion to be of any use in the design of the model, the individual stability derivatives were programmed and expressed in terms of the model physical characteristics. The following sections present the relationships used to compute the required stability derivatives.

A. LONGITUDINAL STABILITY DERIVATIVES

The values of the stability derivatives involved in the longitudinal system of equations were determined using the relationships presented in Ref. 1. These are:

$$C_{m_{\alpha}} = -a(h_n - h) \quad \text{IV-1}$$

$$C_{m_q} = -2a_t \left(\frac{l_t}{\bar{c}} \right) V_H \quad \text{IV-2}$$

$$C_{m_{\dot{\alpha}}} = C_{m_q} \frac{d\epsilon}{d\alpha} \quad \text{IV-3}$$

$$C_{m_{\alpha_t}} = -a_t V_H \quad \text{IV-4}$$

where V_H is the horizontal tail volume ratio given by:

$$V_H = \frac{l_t S_t}{\bar{c} S}$$

From the above equations it is evident that the longitudinal characteristics of the model are very dependent upon the tail length (l_t) and the tail size (S_t), and that by varying the tail area, the longitudinal stability characteristics of the model may vary significantly. For the model under study in this paper:

$$C_{m_\alpha} = -1.144$$

$$C_{m_q} = -14.979$$

$$C_{m_\alpha^*} = -6.231$$

$$C_{m_{\alpha_t}} = -2.9448$$

B. LATERAL STABILITY DERIVATIVES

Unlike the longitudinal case, the expressions for the lateral stability derivatives are very complex and unwieldy to use and program. The equations of Ref. 3 were used to estimate these derivatives and are given in Appendix A. For this particular model, the computer program solution had the following values:

$$C_{l_\beta} = -.1254$$

$$C_{n_\beta} = .07379$$

$$C_{l_p} = -.4381$$

$$C_{n_p} = -.00251$$

$$C_{l_r} = .0437$$

$$C_{n_r} = -.15384$$

$$C_{1_{\xi}} = .69934$$

$$C_{n_{\xi}} = -.001197$$

$$C_{1_{\delta_F}} = .05554$$

$$C_{n_{\delta_F}} = -.23747$$

V. EXPERIMENTAL LONGITUDINAL RESPONSE

A. EQUIPMENT CALIBRATION

With a 4-volt potential across the sensor system, the pitch and roll potentiometers were calibrated using a clinometer, and the yaw potentiometer was calibrated using a protractor with the yaw stopper as a pointer. The resulting calibration curves are presented in this section as Figure V-1. The relationships between control surface deflections and transmitter stick angle were also measured and are presented in Figure V-2.

B. TEST PROCEDURE

The sensor system was energized, after which the wind tunnel was activated and set at the desired airspeed. The model was then trimmed out at a slight nose-up attitude. The horizontal tail surface was deflected and the model response to a step function input was observed and recorded.

Test runs at airspeeds of 50.7, 58.7, 65.4, and 71.7 ft/sec and tail angle changes of 3, 4, 5, and 6 degrees (positive tail angle is with the leading edge up) were accomplished. The graphical results of the model response to a tail deflection of 6 degrees are presented as typical in Figures V-3 through V-6.

From these results, it was possible to determine the values of the damping ratio, ζ , and the undamped natural frequency, ω_n , for the model at the different airspeeds.

Figures V-7 and V-8 clearly illustrate that as the tunnel speed is increased, the value of ω_n remains fairly constant, while the value of ζ decreases. This results in a gradual decline of the damping factor $2\zeta\omega_n$ as the airspeed is increased. This decline in the damping term is a consequence of the increased wing lift at the higher airspeeds which reduces the pivot force resulting in less bearing friction.

Equation III-6 was used to obtain the characteristic equation for the model and the following expression for ω_n and ζ were obtained as functions of the friction coefficient, μ_1 .

$$\omega_n = \sqrt{\frac{-(C_{m_\alpha} - \frac{\mu_1}{\bar{c}} C_{L_\alpha})}{i_B}} \quad V-1$$

$$\zeta = \frac{-(C_{m_\alpha} + C_{m_q})}{2i_B} \sqrt{\frac{i_B}{-C_{m_\alpha} + \frac{\mu_1}{\bar{c}} C_{L_\alpha}}} \quad V-2$$

Figures V-9 and V-10 illustrate these relationships for this model assuming constant μ_1 values.

The experimental results were used to determine the unknown bearing friction coefficient, μ_1 . It was found that a value of 1.75 was required to match the predicted and experimental values of ω_n . Figure V-8 shows the

resulting comparison values for the damping ratio. It is evident that as the tunnel speed increases the experimental and theoretical values show fair agreement.

From the various test runs completed it was found that tunnel speeds around 65 ft/sec exhibited the best characteristics for both visual observation and graphical data acquisition of model response. At velocities greater than 75 ft/sec considerable model buffeting occurs which tends to overshadow any data obtained. Conversely, at tunnel speeds under 50 ft/sec the model response is greatly inhibited by bearing friction, and the resulting heavily damped motion rarely has more than one oscillation.

C. CONCLUSIONS

The model studied has demonstrated that the basic concept of a remotely controlled stability and control wind tunnel test model is a valid one. It has been shown that for the longitudinal case, the damping and frequency of the model may be predicted quite accurately using linear equations of motion.

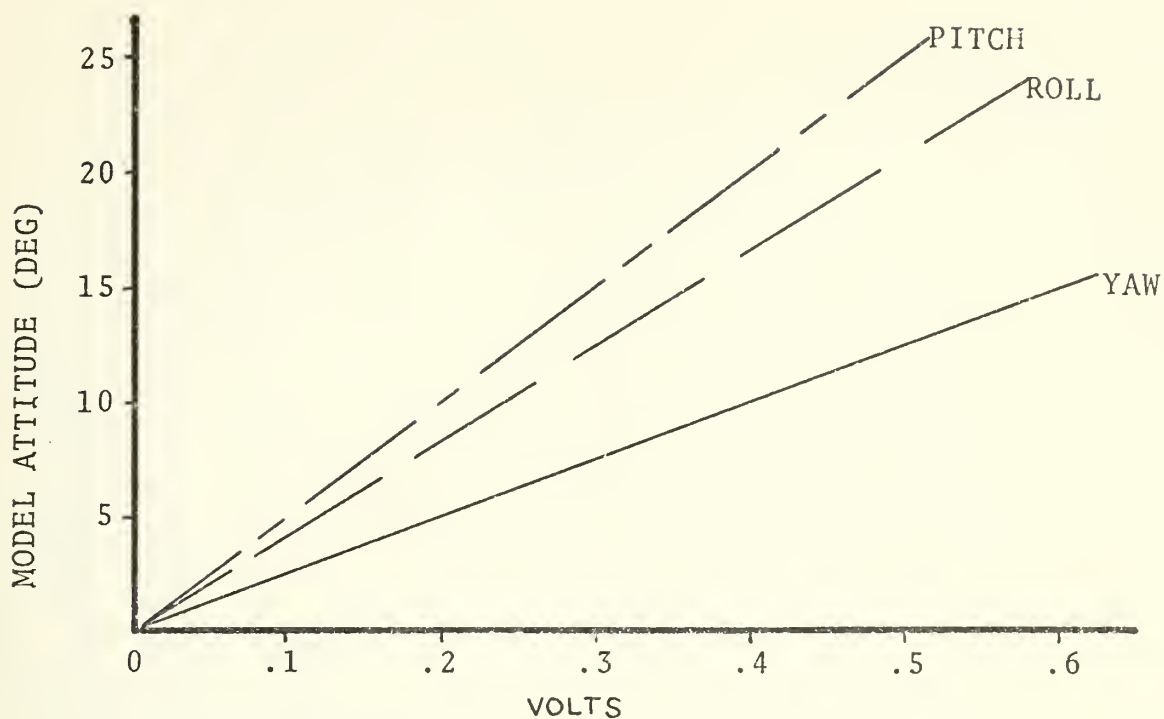


Figure V-1. Potentiometer Calibration Curves

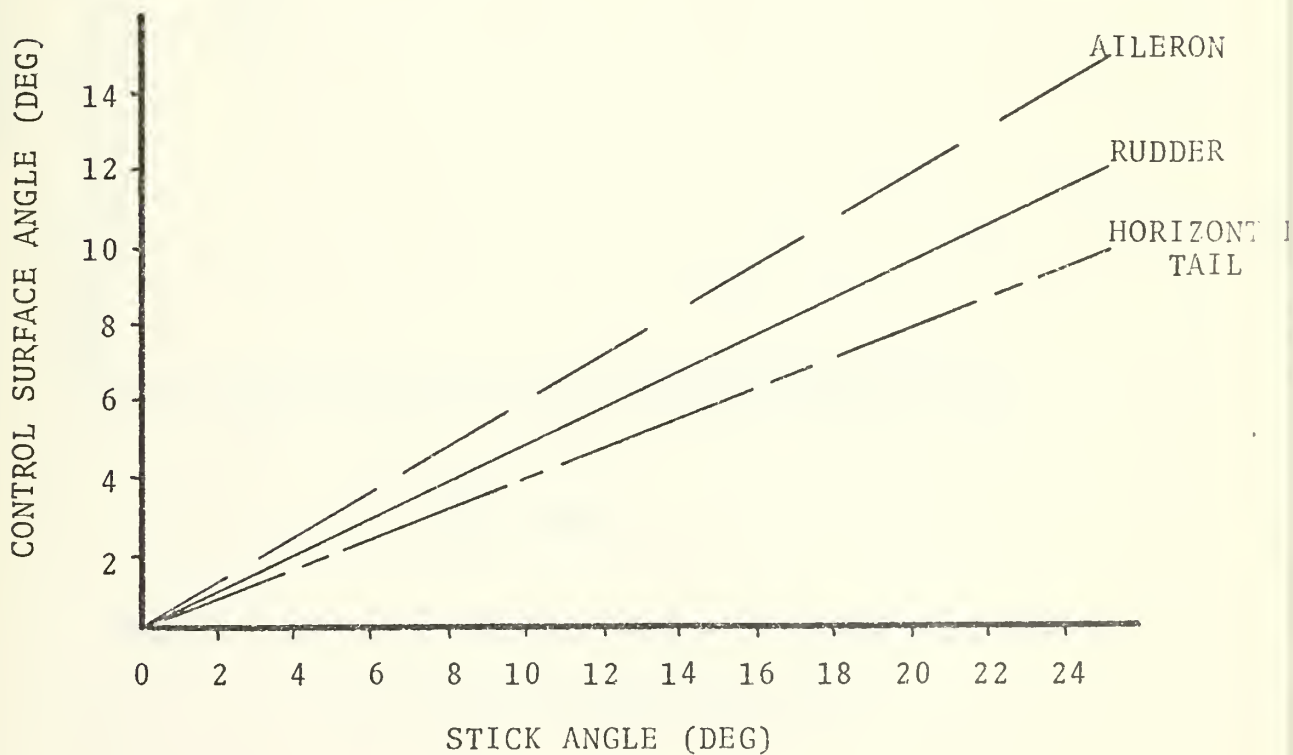


Figure V-2. Control Surface Angle vs. Transmitter Stick Angle

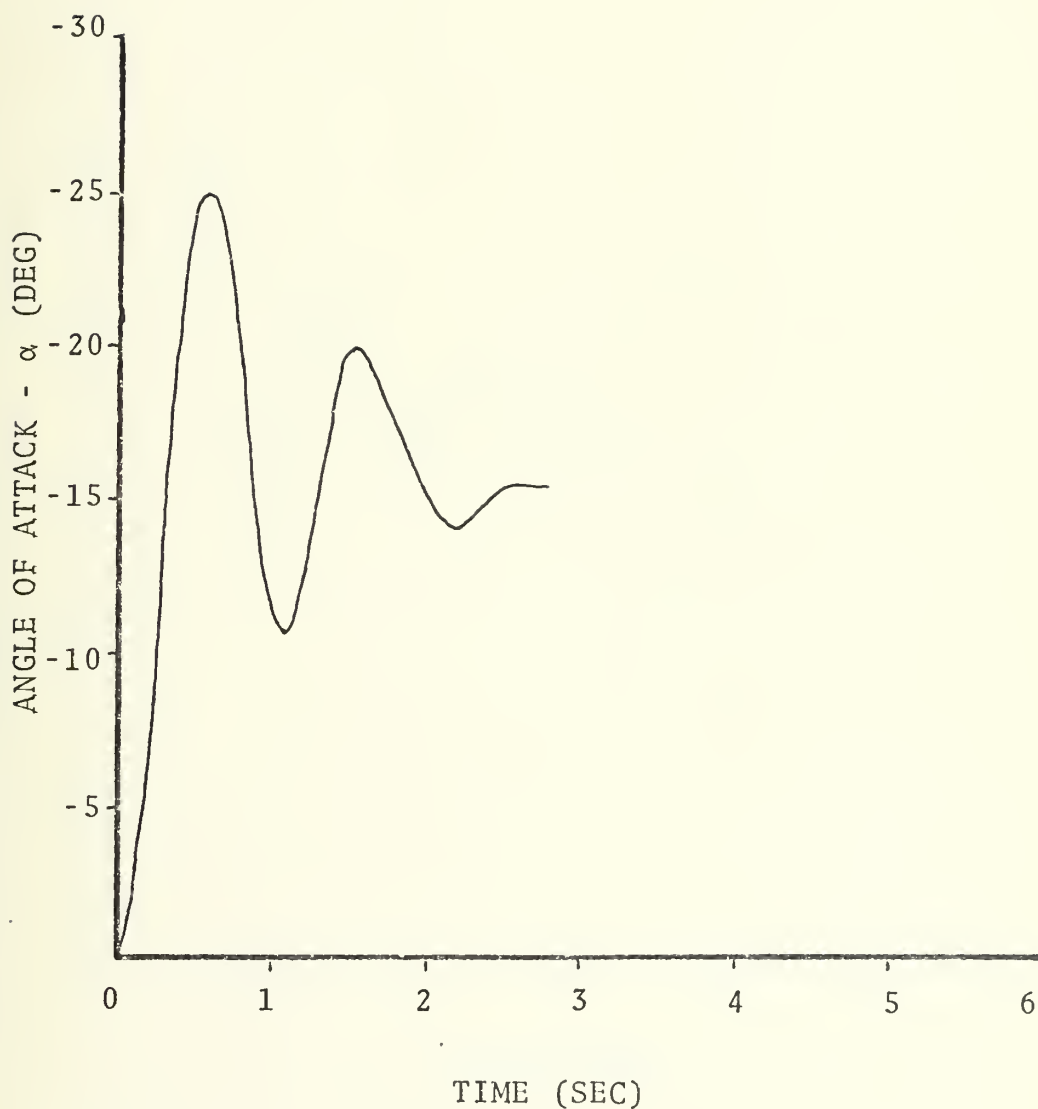


Figure V-3. Experimental Angle of Attack vs. Time

$$V_o = 50.7 \text{ ft/sec}, \Delta\alpha_t = 6^\circ$$

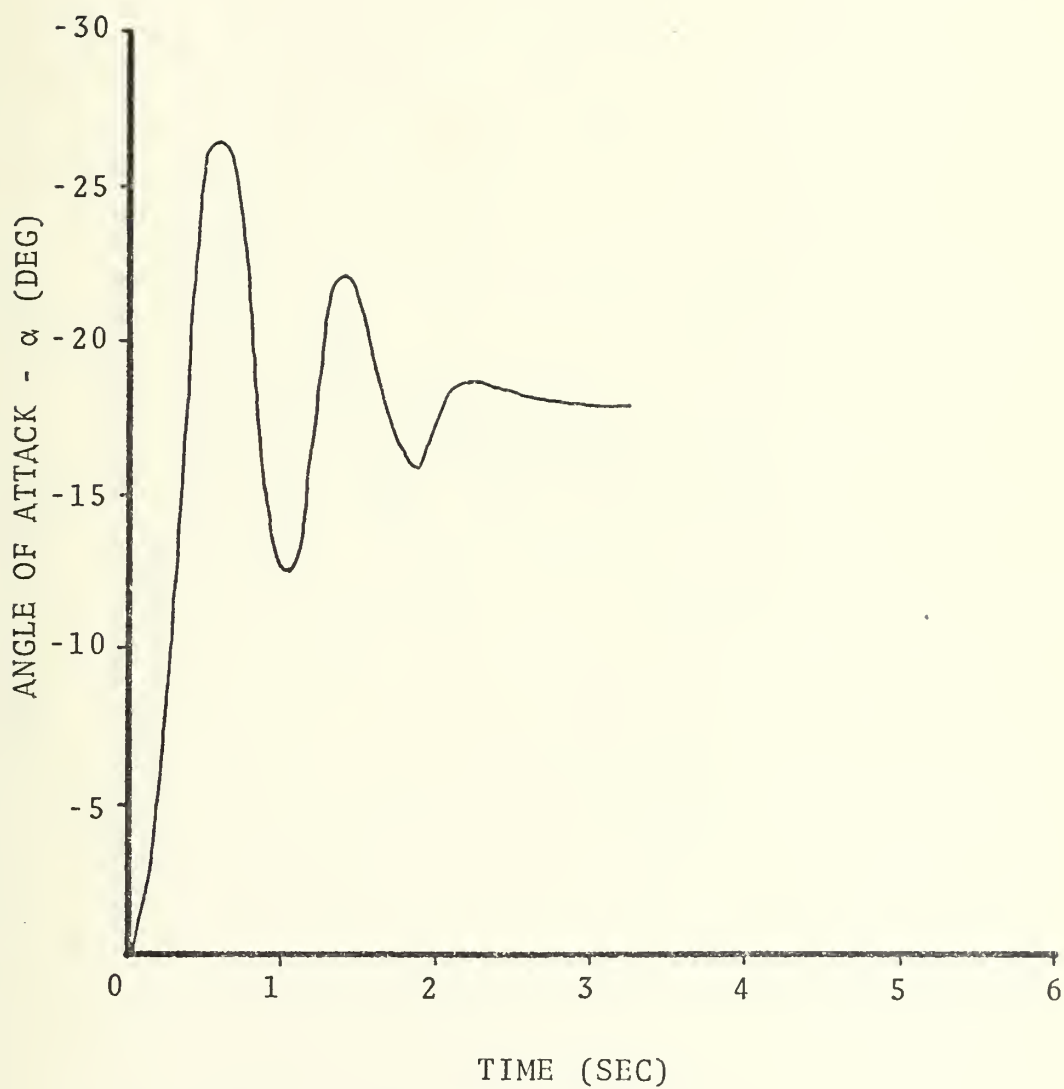


Figure V-4. Experimental Angle of Attack vs. Time

$$V_o = 58.7 \text{ ft/sec}, \Delta\alpha_t = 6^\circ$$

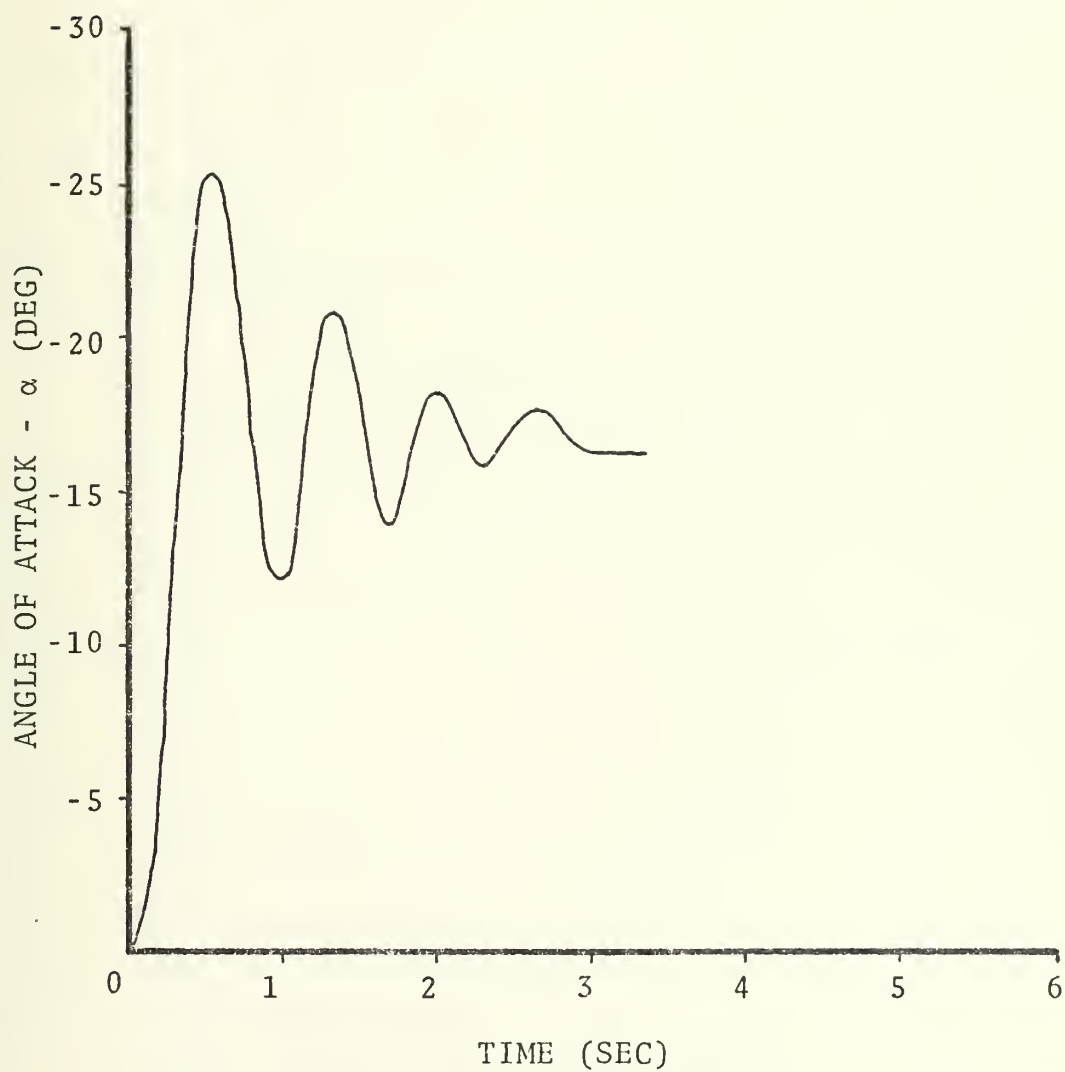


Figure V-5. Experimental Angle of Attack vs. Time

$$V_o = 65.4 \text{ ft/sec}, \Delta\alpha_t = 6^\circ$$

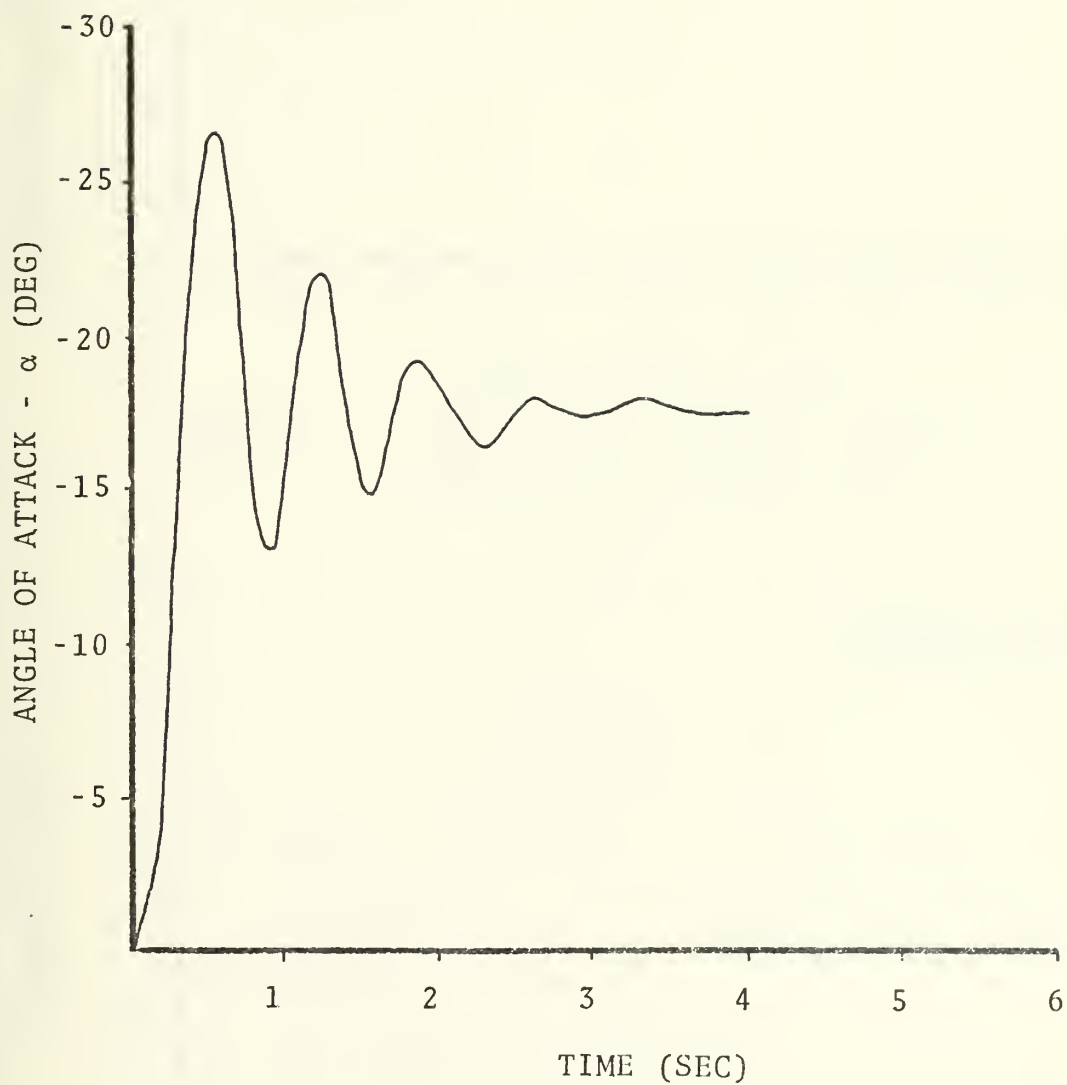


Figure V-6. Experimental Angle of Attack vs. Time

$$V_o = 71.7 \text{ ft/sec}, \Delta\alpha_t = 6^\circ$$

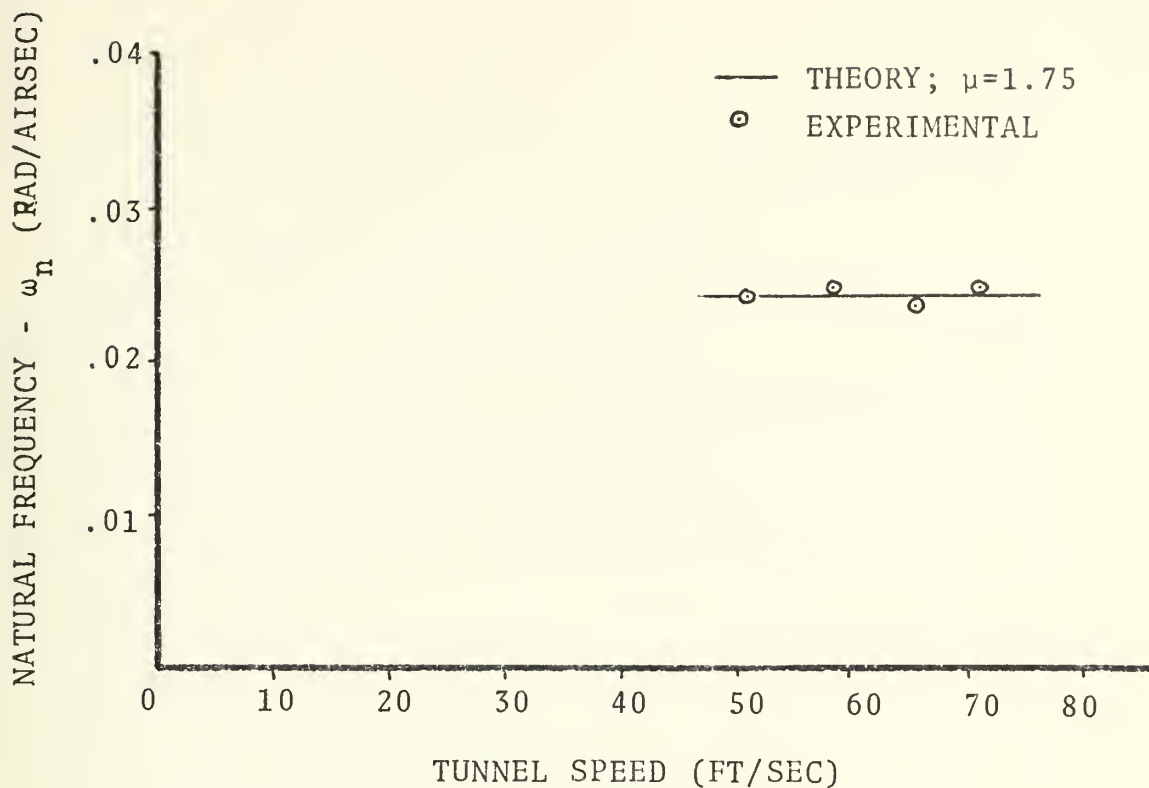


Figure V-7. Longitudinal Undamped Natural Frequency vs. Tunnel Speed

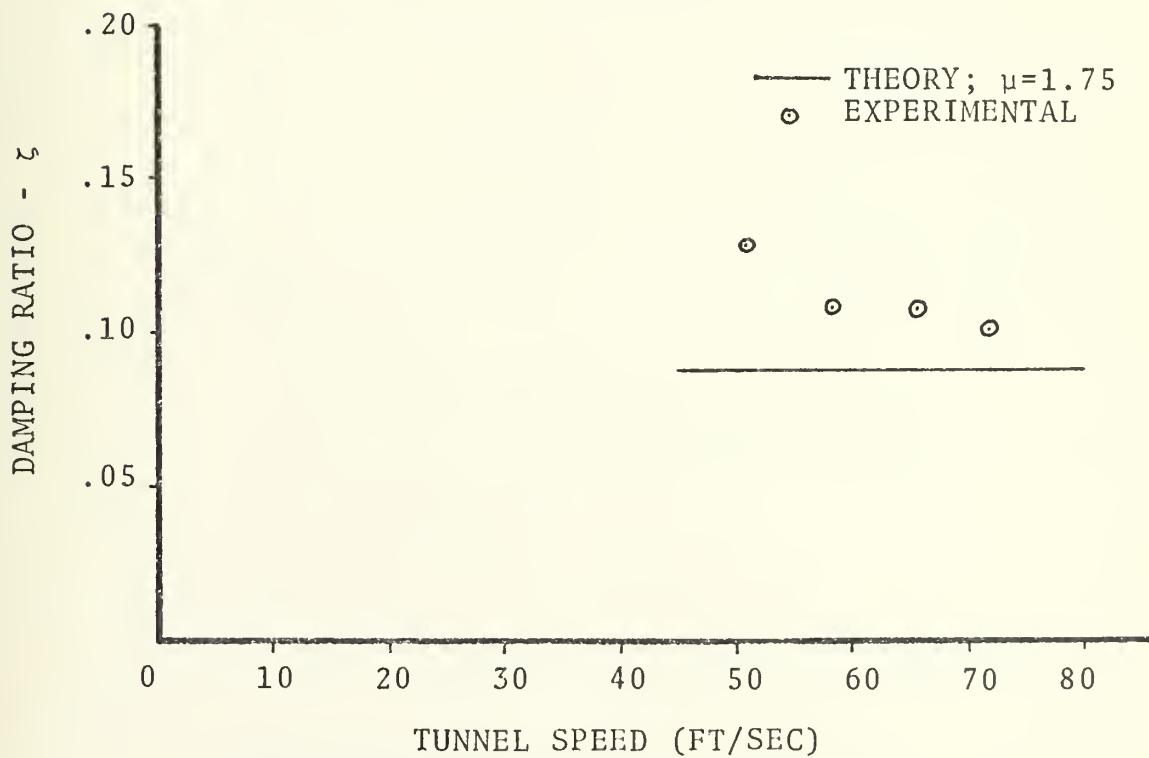


Figure V-8. Longitudinal Damping Ratio vs. Tunnel Speed

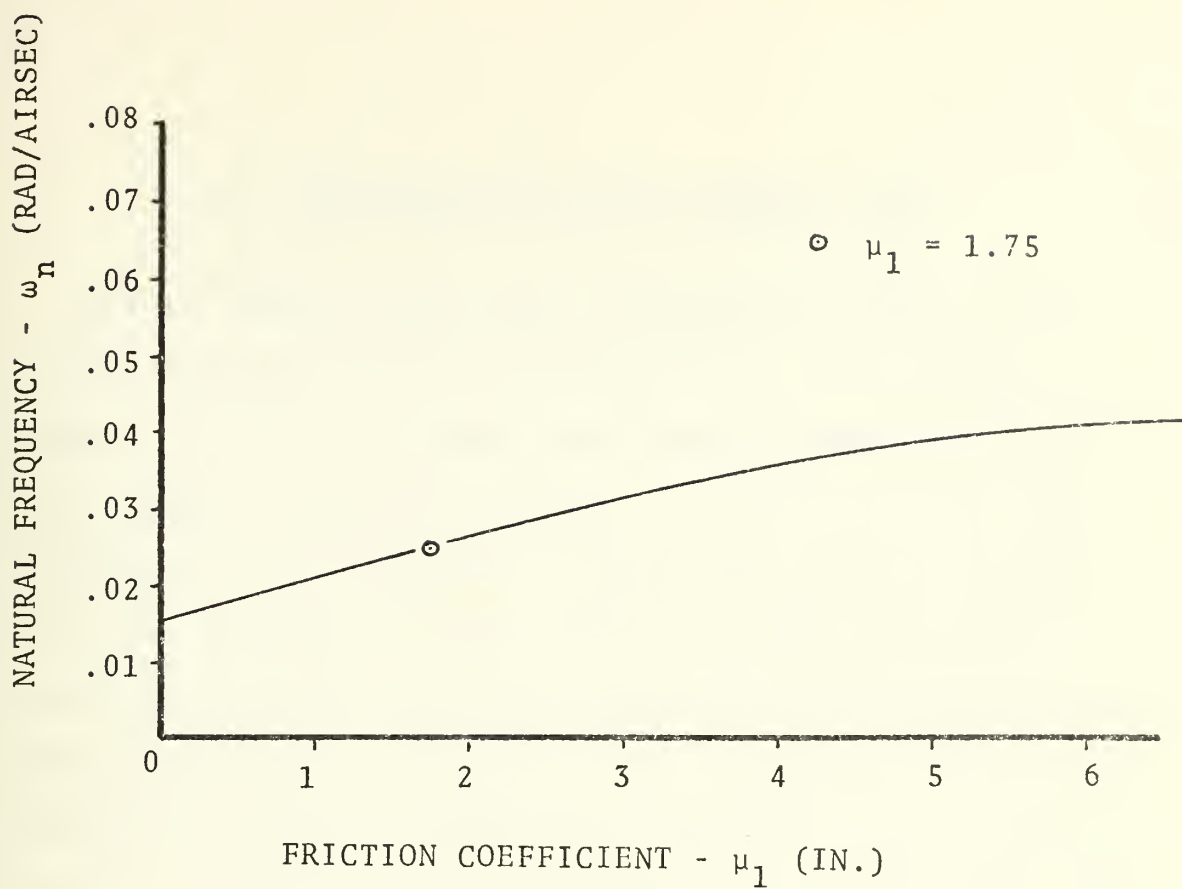


Figure V-9. Predicted Natural Frequency vs. Friction Coefficient

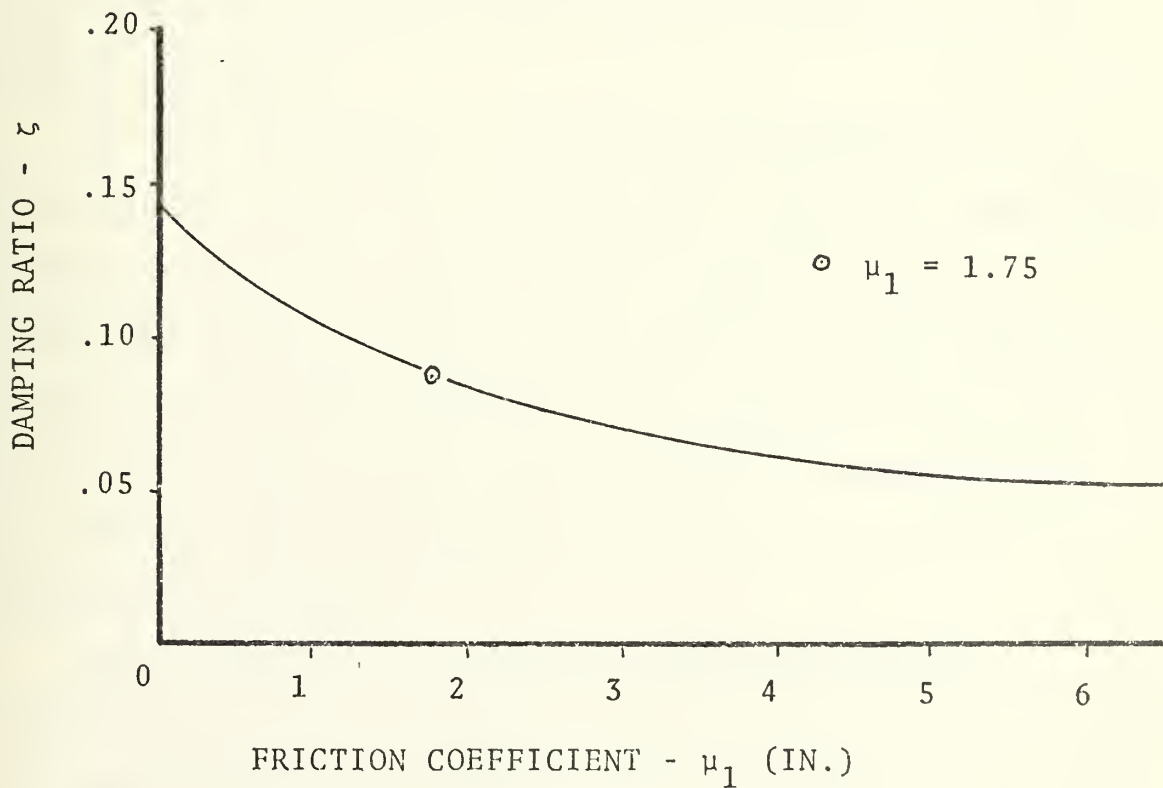


Figure V-10. Predicted Damping Ratio vs. Friction Coefficient

VI. RECOMMENDATIONS FOR FURTHER STUDY

Due to insufficient time, an extensive study of the lateral characteristics of the model was not possible. However, it must be pointed out that the model exhibited very unstable handling qualities in roll. In fact, this instability was detrimental to the point where the model had to be restrained from rolling in order to obtain the degree of accuracy required to satisfactorily determine the model longitudinal characteristics. Further study in this area of the model's response may reveal the reason for, as well as a solution for, this unpredicted instability. The model lateral response should be carefully examined and compared with the theoretical values obtained from the computer program.

The fact that there presently exist difficulties in the roll characteristics of the model should not immediately bring a conclusion that the basic concept of a three-degree-of-freedom stability and control model is impractical. Rather, with continued testing and refinement it is believed that a tool such as this will become an invaluable asset to students actively engaged in the study of aircraft stability and control.

Additionally, it is suggested that the friction coefficient of the main support bearing and yaw potentiometer be experimentally calibrated so that the model response may

be correlated more accurately. In fact, a total redesign of the main support bearing for less friction would greatly simplify and increase the accuracy of the predicted response.

APPENDIX A

LATERAL STABILITY DERIVATIVE EQUATIONS

The following equations for the lateral stability derivatives were obtained from Ref. 3, and, unless otherwise stated, any specific figures mentioned in this appendix are from that reference.

A. VARIATION OF SIDE FORCE COEFFICIENT WITH SIDESLIP ANGLE -

$$C_{y\beta}$$

$$C_{y\beta} = C_{y\beta_W} + C_{y\beta_B} + C_{y\beta_V} \quad A-1$$

The wing contribution to this derivative may be estimated from:

$$C_{y\beta_W} = -.0001 |\Gamma| 57.3 \quad A-2$$

where Γ is the wing geometric dihedral in degrees.

The body contribution may be found by using:

$$C_{y\beta_B} = -2 K_i \left(\frac{S_o}{S} \right) \quad A-3$$

where: K_i = a wing-body interference factor obtained from Fig. 7.1

S_o = the cross sectional area of the fuselage at the point along the body where the flow ceases to be potential.

S = the aircraft reference area.

The vertical tail contribution may be estimated using:

$$C_{y_{\beta_V}} = -k C_{L_{\alpha_V}} \left(1 + \frac{d\sigma}{d\beta}\right) n_V \frac{S_V}{X} \quad A-4$$

where: k = an empirical factor defined in Fig. 7.3

$C_{L_{\alpha_V}}$ = vertical tail lift curve slope

$$\left(1 + \frac{d\sigma}{d\beta}\right) n_V = .724 + 3.06 \frac{S_V/S}{1 + \cos \Lambda_c / 4} + .4 \frac{Z_w}{d} + .009 A$$

S_V = vertical tail area

S = reference area

A = aspect ratio

Z_w = vertical distance from the wing root quarter chord to the fuselage centerline (positive for wing above centerline)

d = fuselage diameter at wing intersection

B. VARIATION OF ROLLING MOMENT COEFFICIENT WITH SIDESLIP
ANGLE - $C_{l_{\beta}}$

$$C_{l_{\beta}} = C_{l_{\beta_{WB}}} + C_{l_{\beta_H}} + C_{l_{\beta_V}} \quad A-5$$

The wing-body contribution may be determined from:

$$C_{1\beta_{WB}} = 57.3 \{ C_{L_{WB}} [(\frac{C_{1\beta}}{C_L})_{\Lambda C/2} K_{M_\Lambda} K_f + (\frac{C_{1\beta}}{C_L})_A] + \Gamma [\frac{C_{1\beta}}{\Gamma} K_{M_\Gamma} + \frac{\Delta C_{1\beta}}{\Gamma}] + (\Delta C_{1\beta})_{Z_w} \}$$

A-6

where: $C_{L_{WB}}$ = wing-body lift coefficient

$(\frac{C_{1\beta}}{C_L})_{\Lambda C/2}$ = wing sweep contribution obtained from Fig. 7.11

K_{M_Λ} = the compressibility correction to sweep from Fig. 7.12

K_f = the fuselage correction factor from Fig. 7.13

$(\frac{C_{1\beta}}{C_L})_A$ = contribution from aspect ratio obtained from Fig. 7.14

Γ = dihedral angle

$\frac{C_{1\beta}}{\Gamma}$ = dihedral effect of the wing from Fig. 7.15

K_{M_Γ} = compressibility correction to dihedral from Fig. 7.16

$\frac{\Delta C_{1\beta}}{\Gamma} = -.0005 A (\frac{d}{b})^2$

b = wing span

d = diameter of fuselage at wing intersection

$(\Delta C_{1\beta})_{Z_w} = \frac{1.2 A}{57.3} (\frac{Z_w}{b}) (\frac{2d}{b})$

Z_w was previously defined for equation A-4.

The horizontal tail contribution is given as:

$$C_{l_{\beta_H}} = C_{l_{\beta_{HB}}} \frac{S_H}{S} \frac{b_H}{b} \quad A-7$$

where: S_H = horizontal tail area

b_H = horizontal tail span

$C_{l_{\beta_{WB}}}$ is obtained by solving equation A-6 using the horizontal tail characteristics vice the wing.

The vertical tail contribution is given by:

$$C_{l_{\beta_V}} = C_{y_{\beta_V}} \frac{(Z_v \cos \alpha - l_v \sin \alpha)}{b} \quad A-8$$

where: $C_{y_{\beta_V}}$ was previously determined using equation A-4

α = angle of attack of the wing

Z_v = absolute distance from fuselage centerline to vertical tail aerodynamic center (a.c.)

l_v = absolute distance from model c.g. to vertical tail a.c.

C. VARIATION OF YAWING MOMENT COEFFICIENT WITH SIDESLIP ANGLE - $C_{n_{\beta}}$

$$C_{n_{\beta}} = C_{n_{\beta_B}} + C_{n_{\beta_V}} \quad A-9$$

The body contribution to equation A-9 is given by:

$$C_{n\beta_B} = -57.3 K_N K_{R_1} \frac{S_{B_S}}{S} \frac{l_B}{b} \quad A-10$$

where: K_N = an empirical factor for body and wing-body effects from Fig. 7.19.

K_{R_1} = Reynolds Number factor for the fuselage from Fig. 7.20

S_{B_S} = body side area

l_B = overall fuselage length

The vertical tail contribution is:

$$C_{n\beta_V} = -C_{y\beta_V} \frac{(l_V \cos \alpha + z_V \sin \alpha)}{b} \quad A-11$$

where all the elements have been previously defined.

D. VARIATION OF ROLLING MOMENT COEFFICIENT WITH ROLL RATE - C_{l_p}

$$C_{l_p} = C_{l_{p_{WB}}} + C_{l_{p_H}} + C_{l_{p_V}} \quad A-12$$

The wing-body contribution to the above equation is:

$$C_{l_{p_{WB}}} \approx C_{l_{p_W}} = \left(\frac{\beta C_{l_p}}{k} \right) \frac{k}{\beta} \quad A-13$$

where: $\left(\frac{\beta C_{l_p}}{k} \right)$ = roll damping parameter from Fig. 8.1

k = the ratio of the average wing section lift
curve slope $C_{l_{\alpha_W}}$ to 2π

$$\beta = \sqrt{1-M^2}$$

The horizontal tail contribution is:

$$C_{l_{p_H}} = .5 (C_{l_p})_H \frac{S_H}{S} \left(\frac{b_H}{b}\right)^2 \quad A-14$$

where: $(C_{l_p})_H$ is obtained by solving equation A-13 for the horizontal tail.

The vertical contribution is estimated using:

$$C_{l_{p_V}} = 2 \left(\frac{z_V}{b}\right)^2 C_{y_{\beta_V}} \quad A-15$$

where all the variables have been previously defined in this appendix.

E. VARIATION OF YAWING MOMENT COEFFICIENT WITH ROLL
RATE - C_{n_p}

$$C_{n_p} = C_{n_{p_W}} + C_{n_{p_V}} \quad A-16$$

The wing contribution is estimated using the following equation.

$$C_{n_{p_W}} = -C_{l_{p_W}} \tan \alpha - [-C_{l_p} \tan \alpha - (\frac{C_{n_p}}{C_L})_{C_L=0} C_L] \quad A-17$$

M

where:

$C_{l_{p_W}}$ is found from equation A-13

α = wing angle of attack

C_L = wing lift coefficient

$$(\frac{C_{n_p}}{C_L})_{C_L=0} = (\frac{A + 4 \cos \Lambda_{c/4}}{AB + 4 \cos \Lambda_{c/4}}) \frac{AB + \frac{1}{2}(AB + \cos \Lambda_{c/4}) \tan^2 \Lambda_{c/4}}{A + \frac{1}{2}(A + \cos \Lambda_{c/4}) \tan^2 \Lambda_{c/4}} (\frac{C_{n_p}}{C_L})_{C_L=0} \quad A-18$$

M

$M=0$

A = aspect ratio

$$B = \sqrt{1 - M^2 \cos^2 \Lambda_{c/4}}$$

M = mach number

$$(\frac{C_{n_p}}{C_L})_{C_L=0} = - \frac{1}{6} \frac{A + 6(A + \cos \Lambda_{c/4}) (\frac{x}{c} \frac{\tan \Lambda_{c/4}}{A} + \frac{\tan^2 \Lambda_{c/4}}{12})}{A + 4 \cos \Lambda_{c/4}} \quad A-19$$

$M=0$

The vertical tail contribution may be found from:

$$C_{n_{p_V}} = - \frac{2}{b} (l_V \cos \alpha + Z_V \sin \alpha) (\frac{Z_V \cos \alpha - l_V \sin \alpha - Z_V}{b}) C_{y_{\beta_V}} \quad A-20$$

where all the elements of A-20 have been previously defined.

F. VARIATION OF ROLLING MOMENT COEFFICIENT WITH YAW
RATE - C_{1_r}

$$C_{1_r} = C_{1_{rW}} + C_{1_{rV}} \quad A-21$$

The wing contribution is given by:

$$C_{1_{rW}} = C_L \left(\frac{C_{1_r}}{C_L} \right)_{C_L=0} + \left(\frac{\Delta C_{1_r}}{\Gamma} \right) \quad A-22$$

M

where: C_L = wing lift coefficient

$$\left(\frac{C_{1_r}}{C_L} \right)_{C_L=0} = \frac{1 + \frac{A(1-B^2)}{2B(AB+2\cos\Lambda_{c/4})} + \frac{AB+2\cos\Lambda_{c/4}}{AB+4\cos\Lambda_{c/4}} \frac{\tan^2\Lambda_{c/4}}{8}}{1 + \frac{A+2\cos\Lambda_{c/4}}{A+4\cos\Lambda_{c/4}} \frac{\tan^2\Lambda_{c/4}}{8}} \left(\frac{C_{1_r}}{C_L} \right)_{C_L=0} \quad A-23$$

$M=0$

$$\left(\frac{C_{1_r}}{C_L} \right)_{C_L=0} \quad \text{is obtained from Fig. 9.1}$$

$M=0$

$$\frac{\Delta C_{1_r}}{\Gamma} = \frac{1}{12} \frac{A \sin\Lambda_{c/4}}{A+4\cos\Lambda_{c/4}}$$

Γ = wing dihedral angle in radians.

The vertical tail contribution is found from:

$$C_{l_{r_V}} = \frac{-2}{b^2} (l_V \cos \alpha + Z_V \sin \alpha) (Z_V \cos \alpha - l_V \sin \alpha) C_{y_{\beta_V}} \quad A-24$$

where all the variables have been previously defined.

G. VARIATION OF YAWING MOMENT COEFFICIENT WITH YAW
RATE - C_{n_r}

$$C_{n_r} = C_{n_{r_W}} + C_{n_{r_V}} \quad A-25$$

The wing contribution is found from:

$$C_{n_{r_W}} = \left(\frac{C_{n_r}}{C_L^2} \right) C_L^2 + \left(\frac{C_{n_r}}{C_{D_o}} \right) C_{D_o} \quad A-26$$

where:

$$\left(\frac{C_{n_r}}{C_L^2} \right) \text{ is found from Fig. 9.4.}$$

$$\left(\frac{C_{n_r}}{C_{D_o}} \right) \text{ is found from Fig. 9.5.}$$

C_{D_o} is the model zero-lift drag coefficient.

The vertical tail contribution may be estimated from:

$$C_{n_{r_V}} = \frac{2}{b^2} (l_V \cos \alpha + Z_V \sin \alpha)^2 C_{y_{\beta_V}} \quad A-27$$

where all the variables have been previously determined.

H. VARIATION OF ROLLING MOMENT COEFFICIENT WITH AILERON DEFLECTION - $C_{l_{\xi}}$

The procedure for determining this coefficient is a rather complex step-by-step process highly dependent upon the exact type of aileron employed. Therefore, it is suggested that Ref. 3 be consulted for the determination of this derivative.

I. VARIATION OF YAWING MOMENT COEFFICIENT WITH AILERON DEFLECTION - $C_{n_{\xi}}$

$$C_{n_{\xi}} = K C_L C_{l_{\xi}} \quad A-28$$

where: K = an empirical factor obtained from Fig. 11.3

C_L = the model steady state lift coefficient

$C_{l_{\xi}}$ = the derivative determined in section H

J. VARIATION OF ROLLING MOMENT COEFFICIENT WITH RUDDER DEFLECTION - $C_{l_{\delta_F}}$

Reference 1 was used to determine the value of this derivative. The following equation was used.

$$C_{l_{\delta_F}} = C_{L_{\alpha_V}} \frac{S_V}{S} \frac{z_V}{b} \quad A-29$$

where: $C_{L_{\alpha_V}}$ = vertical tail lift curve slope

S_V = vertical tail area

K. VARIATION OF YAWING MOMENT COEFFICIENT WITH RUDDER
DEFLECTION - $C_{n\delta_F}$

Reference 1 was used to determine the value of this
derivative also. The following equation was used:

$$C_{n\delta_F} = C_{L\alpha_V} \frac{S_V}{S} \frac{l_V}{b} \quad A-30$$

where all the variables are known.


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**      MODEL CHARACTERISTICS
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TOTAL MODEL PARAMETERS:

TRIM LIFT COEFFICIENT (CL) = 0.0
 MODEL ANGLE OF ATTACK (DEG.) = -5.0982669D-01
 TAIL ANGLE OF ATTACK (DEG) = 2.4765951D 00
 TOTAL LENGTH (IN) = 2.9000000D 01
 TOTAL WEIGHT (LBS) = 4.6997845D 00
 STICK FIXED STATIC MARGIN = 2.2599784D-01
 LIFT CURVE SLOPE (PER RAD.) = 5.0624000D 00

MOMENTS OF INERTIA (IN2-LBS)

I(X) = 6.2023497D 01 I(Y) = 2.6436234D 02
 I(Z) = 3.1432216D 02 I(XZ) = 1.0348581D 01

MODEL C.G. WITH RESPECT TO REFERENCE AXIS ORIGIN (IN)

XBAR = 4.7068655D-02 YBAR = -3.1916357D-03
 ZBAR = -1.6414961D-02

WING AND TAIL SURFACE PARAMETERS:

-WING CHARACTERISTICS-

ROOT CHORD (IN) =	6.1500000D 00
TIP CHORD (IN) =	1.8500000D 00
WING SPAN (IN) =	3.0000000D 01
WING AREA (IN2) =	1.2000000D 02
L.E. SWEEP (DEG) =	2.3500000D 01
DIHEDRAL ANGLE (DEG) =	6.0000000D 00
LIFT CURVE SLOPE (PER RAD) =	4.5309000D 00

-HORIZONTAL TAIL CHARACTERISTICS-

ROOT CHORD (IN) =	3.9375000D 00
TIP CHORD (IN) =	9.3000000D-01
TAIL SPAN (IN) =	1.3125000D 01
TAIL AREA (IN2) =	3.1100000D 01
L.E. SWEEP (DEG) =	2.5200000D 01
DIHEDRAL ANGLE (DEG) =	0.0
LIFT CURVE SLOPE (PER RAD) =	4.0211000D 00

-VERTICAL TAIL CHARACTERISTICS-

ROOT CHORD (IN) =	6.5000000D 00
TIP CHORD (IN) =	1.5000000D 00
TAIL HEIGHT (IN) =	6.3750000D 00
TAIL AREA (IN2) =	2.6000000D 01
L.E. SWEEP (DEG) =	5.0000000D 01
LIFT CURVE SLOPE (PER RAD) =	2.9891000D 00

LONGITUDINAL STABILITY DERIVATIVES:

CM- = VARIATION OF PITCHING MOMENT COEFFICIENT
WITH -

A = ANGLE OF ATTACK (ALFA) Q = PITCH RATE

A(DOT) = VARIATION OF ALFA WITH RESPECT TO TIME

CMQ = -1.4979070D 01 CMA = -1.1440915D 00

CMA(DOT) = -6.2308383D 00 CMA(TAIL) = -2.9448839D 00

ELEMENTS OF STATE VARIABLE MATRIX A ARE:

-4.4970234D-03	-2.4257559D-04
1.0000000D 00	0.0

ELEMENTS OF THE B VECTOR ARE:

-6.2438798D-04
0.0

LATERAL STABILITY DERIVATIVES:

CL- = VARIATION OF ROLLING MOMENT COEFFICIENT
WITH -

CN- = VARIATION OF YAWING MOMENT COEFFICIENT
WITH -

B = SIDESLIP ANGLE (BETA)

P = ROLL RATE

R = YAW RATE

DA = AILERON DEFLECTION

DF = RUDDER DEFLECTION

CLB = -1.2537315D-01

CNB = 7.3790651D-02

CLP = -4.3806752D-01

CNP = -2.5131716D-03

CLR = 4.3663509D-02

CNR = -1.5384192D-01

CLDA = 6.9936720D-01

CNDA = -1.1977032D-03

CLDF = 5.5541463D-02

CNDF = -2.3746739D-01

ELEMENTS OF STATE VARIABLE MATRIX A ARE:

-1.2747884D-01	1.1230146D-02	-3.5770197D-02	0.0
-4.3405394D-03	-8.4139888D-03	3.0354552D-03	0.0
0.0	-1.0000000D 00	0.0	0.0
1.0000000D 00	0.0	0.0	0.0

ELEMENTS OF THE B MATRIX ARE:

2.0346785D-01	1.3884961D-02
6.6304872D-03	-1.3101244D-02
0.0	0.0
0.0	0.0

THIS PROGRAM COMPUTES THE MODEL LONGITUDINAL AND LATERAL STABILITY DERIVATIVES AS WELL AS THE MOMENTS OF INERTIA. IT THEN USES THESE VALUES TO DETERMINE THE ELEMENTS OF THE 'A', AND 'B' MATRICES, FOR THE FRICTIONLESS CASE, WHICH ARE USED IN THE STATE REPRESENTATION OF THE EQUATIONS OF MOTION.

UNLESS OTHERWISE STATED, ALL LENGTH QUANTITIES ARE IN UNITS OF INCHES AND ALL AREA QUANTITIES ARE IN UNITS OF SQUARE INCHES. ALL FIGURES REFERRED TO IN THE SECTION DESCRIBING THE REQUIRED INPUT CARDS MAY BE FOUND IN REF. 3 OF THIS THESIS. THIS PROGRAM IS NOT APPLICABLE FOR A MODEL HAVING A WING WITH GEOMETRIC TWIST AND SHOULD NOT BE USED IN SUCH A CASE.

UNLESS OTHERWISE SPECIFIED, ALL INPUT CARDS ARE OF THE FORMAT XF10.7, WHERE 'X' IS THE NUMBER OF REAL CONSTANTS TO BE PUNCHED IN EACH CARD. A DESCRIPTION OF THE INPUT QUANTITIES REQUIRED FOR THIS PROGRAM FOLLOWS.

CARD # 1: FL - OVERALL FUSELAGE LENGTH
COL. 1) CN - ABSOLUTE DISTANCE FROM NOSE OF FUSELAGE TO C.G.
21) T - FUSELAGE SKIN THICKNESS

CARD # 2: DENS - DENSITY OF SKIN MATERIAL (LB/FT**3)
COL. 1) DENW - DENSITY OF WING MATERIAL (LB/FT**3)
21) DENT - DENSITY OF HORIZONTAL TAIL MATERIAL (LB/FT**3)
31) DENV - DENSITY OF VERTICAL TAIL MATERIAL (LB/FT**3)

CARD # 3: D(1) - INSIDE DIAMETER OF FUSELAGE AT NOSE
COL. 1)

CARD # 4: D(2) - INSIDE DIAMETER OF FUSELAGE AT (FL)/20 IN. AFT OF THE NOSE
COL. 1)

CARD # 5: D(3) - INSIDE DIAMETER OF FUSELAGE AT (2*FL)/20 IN. AFT OF THE NOSE
COL. 1)

" " " " " "

" " " " " " " "

CARD # (Y+2):
COL. 1) D(Y) - INSIDE DIAMETER OF FUSELAGE AT (Y-1)*FL/20 IN AFT OF NOSE

" " " " " " " "

CARD #23:
COL. 1) D(21) - INSIDE DIAMETER AT THE TAIL OF THE FUSELAGE

CARD #24:
COL. 1) RT - AVERAGE THICKNESS OF MAIN WING AT THE FUSELAGE
11) TT - AVERAGE THICKNESS OF MAIN WING AT TIP
21) CT - WING TIP CHORD
31) CR - WING CHORD AT FUSELAGE
41) CR1 - WING ROOT CHORD

CARD #25:
COL. 1) RMPT - LOCATION ALONG THE X-AXIS OF THE MIDPOINT OF THE WING CHORD
11) TMPT - AT THE FUSELAGE JUNCTION (NEGATIVE IF AFT OF C.G.)
21) WS - LOCATION ALONG X-AXIS OF WING TIP CHORD MIDPOINT (NEG. IF
31) DISTW - LENGTH OF EXPOSED WING SEMI-SPAN
- LENGTH OF WING SEMI-SPAN COVERED BY FUSELAGE

CARD #26:
COL. 1) GAMAW - DIHEDRAL ANGLE OF THE WING (DEG)
11) SWLEW - LEADING EDGE SWEEP OF THE WING (DEG)

CARD #27:
COL. 1) CRT - CHORD OF THE HORIZONTAL TAIL AT THE FUSELAGE
11) CTT - HORIZONTAL TAIL TIP CHORD
21) TRI - AVERAGE THICKNESS OF THE HORIZONTAL TAIL AT THE FUSELAGE
31) TTT - AVERAGE THICKNESS OF THE HORIZONTAL TAIL AT THE TIP
41) CRT1 - HORIZONTAL TAIL ROOT CHORD

CARD #28:
 COL. 1) DIST - LENGTH OF HORIZ. TAIL SEMI-SPAN COVERED BY THE FUSELAGE
 COL. 11) TMPTR- LOCATION ALONG X-AXIS OF THE MIDPT. OF THE HORIZ. TAIL CHORD
 21) TMPPT- LOCATION ALONG X-AXIS OF HORIZ. TAIL TIP CHORD MIDPT.

CARD #29:
 COL. 1) GAMAT - HORIZ. TAIL DIHEDRAL ANGLE (DEG.)
 COL. 11) TS - LENGTH OF EXPOSED HORIZ. TAIL SEMI-SPAN
 21) SWLET - HORIZ. TAIL LEADING EDGE SWEEP (DEG)

CARD #30:
 COL. 1) VRC - CHORD OF VERTICAL TAIL AT THE FUSELAGE
 COL. 11) VTC - VERT. TAIL TIP CHORD
 21) VRT - AVERAGE THICKNESS OF VERT. TAIL CHORD AT FUSELAGE
 31) VTT - AVERAGE THICKNESS OF VERT. TAIL CHORD AT TIP
 41) VRC1- VERT. TAIL ROOT CHORD

CARD #31:
 COL. 1) VDIS - PORTION OF THE TAIL HEIGHT COVERED BY FUSELAGE
 COL. 11) VMPTR- LOCATION ALONG X-AXIS OF THE MIDPT. OF THE VERT. TAIL CHORD
 21) VMPTT- LOCATION ALONG X-AXIS OF VERT. TAIL TIP CHORD MIDPT.
 31) VB - EXPOSED VERTICAL TAIL HEIGHT

CARD #32:
 COL. 1) VS - VERT. TAIL AREA INCLUDING PORTION COVERED BY FUSELAGE
 COL. 11) CLAV - VERT. TAIL LIFT CURVE SLOPE (PER RAD.)
 21) SWLEV- VERT. TAIL LEADING EDGE SWEEP (DEG)

CARD #33: ***NOTE: FORMAT I2***
 COL. 1&2) N - NUMBER OF INTERNAL LUMPED MASSES (AN INTEGER)

CARD #34A:
 COL. 1) W - THE WEIGHT OF THE FIRST LUMPED MASS (LBS.)
 COL. 11) X - THE X LOCATION OF THE PARTICULAR MASS
 21) Y - THE Y LOCATION OF THE MASS
 31) Z - THE Z LOCATION OF THE MASS

CARD #34B:
 COL. 1) W - THE WEIGHT OF THE SECOND MASS (LBS.)

11) X - THE X LOCATION OF THE SECOND MASS
 21) Y - THE Y LOCATION OF THE SECOND MASS
 31) Z - THE Z LOCATION OF THE SECOND MASS

" " " " " "

" " " " " "

CARD #34N:
 COL. 1) W - THE WEIGHT OF THE NTH MASS (LBS.)
 11) X - THE X LOCATION OF THE NTH MASS
 21) Y - THE Y LOCATION OF THE NTH MASS
 31) Z - THE Z LOCATION OF THE NTH MASS

CARD #35: PUT * IN THE FIRST 20 COLS.

CARD #36:
 COL. 1) H - C.G. POSITION, FRACTION OF M.A.C.
 11) CLTRM- DESIRED TRIM LIFT COEFFICIENT

CARD #37:
 COL. 1) ST - HORIZ. TAIL AREA INCLUDING PORTION COVERED BY FUSELAGE

CARD #38:
 COL. 1) TL - ABSOLUTE DISTANCE FROM C.G. TO HORIZ. TAIL A.C.
 11) TH - VERT. DISTANCE BETWEEN HORIZ. TAIL ROOT CHORD AND FUSELAGE CENTERLINE (POSITIVE FOR TAIL BELOW CENTERLINE)
 21) CLAT - HORIZ. TAIL LIFT CURVE SLOPE (PER RAD.)
 31) CMOWB- ZERO LIFT PITCHING MOMENT OF THE WING-BODY COMBINATION

CARD #39:
 COL. 1) WA - WING AREA INCLUDING PORTION COVERED BY FUSELAGE
 11) WB - WING SPAN
 21) HNWB - NEUTRAL POINT OF WING-BODY COMBINATION, FRACTION OF M.A.C.
 31) SWLEW- WING LEADING EDGE SWEEP (DEG.)

CARD #40:
 COL. 1) CR1 - WING ROOT CHORD
 11) CT - WING TIP CHORD

21) GAMAW- WING DIHEDRAL ANGLE (DEG.)
 31) ALFAZL- WING ANGLE OF ATTACK FOR ZERO LIFT (DEG.)

CARD #41:
 COL. 11) CLAW- WING LIFT CURVE SLOPE (PER RAD.)
 11) CLAWS- WING SECTION LIFT CURVE SLOPE (PER RAD.)
 21) CLAM- MODEL LIFT CURVE SLOPE (PER RAD.)
 31) CLAWB- WING-BODY LIFT CURVE SLOPE (PER RAD.)

CARD #42:
 COL. 1) CLBCLS - WING SWEEP CONTRIBUTION OF VARIATION OF ROLLING MOMENT
 OF REF. COEFFICIENT WITH SIDESLIP ANGLE, OBTAINED FROM FIG. 7.11
 11) SKM - COMPRESSIBILITY CORRECTION TO SWEEP OBTAINED FROM FIG. 7.12
 21) FK - FUSELAGE CORRECTION FACTOR OBTAINED FROM FIG. 7.13
 31) CLBCLA - ASPECT RATIO CONTRIBUTION FROM FIG. 7.14
 41) CLBG - WING DIHEDRAL EFFECT FROM FIG. 7.15
 51) GKM - COMPRESSIBILITY CORRECTION TO DIHEDRAL FROM FIG. 7.16

CARD #43:
 COL. 1) H - C.G. POSITION, FRACTION OF M.A.C.
 11) ZW - VERT. DISTANCE FROM WING ROOT TO FUSELAGE CENTERLINE (POSITIVE
 FOR WING ROOT BELOW CENTERLINE)

CARD #44:
 COL. 1) HCBCLS - ..
 11) HSKM - ..
 21) HFKE - ..
 31) HCBCLA - ..
 41) HCLBG - ..
 51) HGKM - ..
 SAME CONSTANTS AS DESCRIBED IN CARD #42 BUT
 EVALUATED FOR THE HORIZONTAL TAIL

CARD #45:
 COL. 1) ZWH - VERT. DISTANCE FROM HORIZ. TAIL ROOT TO FUSELAGE CENTERLINE
 (POSITIVE FOR TAIL ROOT BELOW CENTERLINE)

CARD #46:
 COL. 1) CONSTV - EMPIRICAL FACTOR FOR ESTIMATING THE CONTRIBUTION TO
 SIDESLIP FOR SINGLE VERT. TAIL FROM FIG. 7.3
 11) CLAV - VERT. TAIL LIFT CURVE SLOPE (PER RAD.)
 21) VS - VERT. TAIL AREA INCLUDING PORTION COVERED BY FUSELAGE

CARD #47:
 COL. 1) ZV - ABSOLUTE DISTANCE FROM FUSELAGE CENTERLINE TO VERT. TAIL A.C.
 11) VL - ABSOLUTE DISTANCE FROM C.G. TO VERT. TAIL A.C.

CARD #48:
 COL. 1) XKN - WING-BODY INTERFERENCE FACTOR FROM FIG. 7.19
 11) XKRL - EFFECT OF FUSELAGE REYNOLDS NUMBER FROM FIG. 7.20
 21) SBS - BODY SIDE AREA
 31) FL - TOTAL FUSELAGE LENGTH

CARD #49:
 COL. 1) XBCLPK - ROLL DAMPING PARAMETER FROM FIG. 8.1
 11) CLAWS - WING SECTION LIFT CURVE SLOPE (PER RAD.)
 21) HBCLPK - ROLL DAMPING PARAMETER FOR HORIZONTAL TAIL FROM FIG. 8.1
 31) CLATS - LIFT CURVE SLOPE OF HORIZONTAL TAIL SECTION (PER RAD.)

CARD #50:
 COL. 1) XBARI - DISTANCE FROM MODEL C.G. TO MODEL A.C. (POSITIVE WHEN
 A.C. IS AFT OF C.G.)

CARD #51:
 COL. 1) XCLRCL - SLOPE OF THE LOW SPEED ROLLING MOMENT DUE TO YAWING
 AT ZERO LIFT OBTAINED FROM FIG. 9.1

CARD #52:
 COL. 1) XCNRCL - LOW SPEED DRAG DUE TO LIFT YAW-DAMPING PARAMETER OBTAINED
 FROM FIG. 9.4
 11) XCNRCD - LOW SPEED DRAG YAW-DAMPING PARAMETER FROM FIG. 9.5
 21) CDO - ZERO-LIFT DRAG COEFFICIENT OF THE MODEL

CARD #53:
 COL. 1) BCPLDK -AILERON ROLLING MOMENT PARAMETER FROM FIG. 11.1
 11) CLDTD - EMPIRICAL CORRECTION FOR LIFT EFFECTIVENESS OF PLAIN
 TRAILING EDGE FLAPS FROM FIG. 10.6
 21) CLDT - LIFT EFFECTIVENESS OF PLAIN TRAILING EDGE FLAPS FROM
 FIGURE 10.5
 31) CK - CORRELATION CONSTANT FOR DETERMINING YAWING MOMENT DUE
 TO AILERON DEFLECTIONS FROM FIG. 11.3

***** THIS ENDS THE REQUIRED DATA INPUTS *****


```

10      IMPLICIT REAL*8(A-H,O-Z)
11      CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
12      CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
13      C      COMPUTATION OF MODEL C.G. AND MOMENTS OF INERTIA C
14      CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
15      C      DIMENSION D(30), RAD(30)
16      C      DIMENSION WM(30), XM(30), YM(30), ZM(30)
17      C      COMPUTATION OF C.G. AND MOMENTS OF INERTIA DUE TO THE FUSELAGE
18      C
19      C      READ (5,170) FL,CN,T
20      C      READ (5,160) DENSK,DENW,DENT,DENV
21      C      WEIGHF = 0.000
22      C      PRODFV = 0.000
23      C      PRODFZ = 0.000
24      C      A1 = 0.000
25      C      B1 = 0.000
26      C      C1 = 0.000
27      C      E1 = 0.000
28      C      SL = FL/20.00
29      C      SS = SL/5.00
30      C      READ (5,130) D(1)
31      C      RAD(1) = (D(1)+T)/2.00
32      C
33      C      DO 40 I=2,21
34      C      READ (5,130) D(I)
35      C      RAD(I) = (D(I)+T)/2.00
36      C      SLOPE = (RAD(I)-RAD(I-1))/SL
37      C
38      C      DO 30 J=1,9,2
39      C      SUBINT = DFL0AT(J)*SS/2.00
40      C      UNIT = SUBINT+(I-2)*SL
41      C      X = CN-UNIT
42      C      R = SUBINT*SLOPE+RAD(I-1)
43      C      SW = R*4.00/57.2957700
44      C      DELMF = SS*SW*T*DENSK/(12**3)
45      C

```


C C

```

DC 20 K=2,358,4
ALFA = DFL0AT(K)/57.29577D0
Z = R*DCOS(ALFA)
Y = -R*DSIN(ALFA)
WEIGHF = DELMF+WEIFGHF
PRODFV = DELMF*X+PRODF
PRODFZ = DELMF*Y+PRODFV
A1 = DELMF*(Y**2+Z**2)+A1
B1 = DELMF*(X**2+Z**2)+B1
C1 = DELMF*(X**2+Y**2)+C1
E1 = DELMF*X*Z+E1
20 CONTINUE

```

C C C

30 CONTINUE

C C C

40 CONTINUE

C C C C C

COMPUTATION OF C.G. AND MOMENTS OF INERTIA DUE TO THE WINGS

```

READ (5,190) RT,TT,CT,CR,CRI
READ (5,160) RMPT,TMPT,WS,DISTW
READ (5,150) GAMAW,SWLEW
WEIGHW = 0.0D0
PRODW = 0.0D0
PRODWZ = 0.0D0
A2 = 0.0D0
B2 = 0.0D0
C2 = 0.0D0
E2 = 0.0D0
GAMA = GAMAW/57.29577D0
THCKR = 1.D0-TT/RT
TAPR = 1.D0-CT/CR
WL = WS/50.D0
SLOPE = (TMPT-RMPT)/WS

```

C C C

```

DO 50 N=1,99,2
UNIT = DFL0AT(N)*WL/2.D0

```


940
950
960
970
980
990
1000
1010
1020
1030
1040
1050
1060
1070
1080
1090
1100
1110
1120
1130
1140
1150
1160
1170
1180
1190
1200
1210
1220
1230
1240
1250
1260
1270
1280
1290
1300
1310
1320
1330
1340
1350
1360
1370
1380
1390
1400

```
HYP = DISTW*UNIT
CORD = CR*(1.00-TAPR*UNIT/WS)
THICK = RT*(1.00-THCKR*UNIT/WS)
VOLW = WL*CORD*THICK/(12**3)
DELMW = VOLW*DENMPT
X = SLOPE*UNIT+RMPT
Y = HYP*DCOS(GAMA)
Z = -HYP*DSIN(GAMA)
WEIGHW = DELMW*WEIGHW
PRODWZ = DELMW*X+PRODWZ
A2 = DELMW*(Y**2+Z**2)+A2
B2 = DELMW*(X**2+Z**2)+B2
C2 = DELMW*(X**2+Y**2)+C2
E2 = DELMW*X*Z+E2
50 CONTINUE
```

CC

```
WEIGHW = 2.00*WEIGHW
PRODWZ = 2.00*PRODWZ
A2 = 2.00*A2
B2 = 2.00*B2
C2 = 2.00*C2
E2 = 2.00*E2
```

CC

COMPUTATION OF C.G. AND MOMENTS OF INERTIA DUE TO THE HORIZONTAL TAIL

```
READ (5,190) CRT,CTI,TRT,TTT,CRTI
READ (5,170) DIST,TMPTR,TMPTT
READ (5,170) GAMAT,TS,SWLET
TB = 2.00*(DIST+TS)
GAMAT = GAMAT/57.29577D0
WEIGHH = 0.0D0
PRCDH = 0.0D0
PRODHZ = 0.0D0
A3 = 0.0D0
B3 = 0.0D0
C3 = 0.0D0
E3 = 0.0D0
THICKR = 1.00-TTT/TRT
TAPR = 1.00-CTT/CRT
SLOPE = (TMPTT-TMPTR)/TS
TL = TS/25.0D0
```

CC


```

DC 60 N=1,49,2
UNIT = DFL0AT(N)*TL/2.D0
HYP = DIST+UNIT
TCORD = CRT*(1.D0-TAPR*UNIT/IS)
TTHIK = TRT*(1.D0-THICKR*UNIT/TS)
VOLT = TL*TCORD*TTHIK/(12**3)
DELMT = VOLT*DEMENT
X = SLOPE*UNIT+TMPTR
Y = HYP*DCOS(GAMAT)
Z = -HYP*DSIN(GAMAT)
WEIGHH = DELMT*WEIGHH
PRODHZ = DELMT*X+PRODH
PRODHZ = DELMT*Z+PRODHZ
A3 = DELMT*(Y**2+Z**2)+A3
B3 = DELMT*(X**2+Z**2)+B3
C3 = DELMT*(X**2+Y**2)+C3
E3 = DELMT*X*Z+E3
60 CONTINUE

```

CC

```

WEIGHH = 2.D0*WEIGHH
PRODHZ = 2.D0*PRODH
PRODHZ = 2.D0*PRODHZ
A3 = 2.D0*A3
B3 = 2.D0*B3
C3 = 2.D0*C3
E3 = 2.D0*E3

```

CC

COMPUTATION OF C.G. AND MOMENTS OF INERTIA DUE TO THE VERTICLE TAIL

```

READ (5,190) VRC,VTC,VRT,VTT,VRC1
READ (5,160) VDIS,VMPTR,VMPTT,VB
READ (5,170) VS,CLAV,SWLEV
VDIS = DABS(VDIS)
WEIGHV = 0.000
PRODV = 0.000
PRODVZ = 0.000
A4 = 0.000
B4 = 0.000
C4 = 0.000
E4 = 0.000
THICKR = 1.D0-VTT/VRT
TAPR = 1.D0-VTC/VRC
SLOPE = (VMPTT-VMPTR)/VB
VW = VB/25.D0

```

CC

1420
1430
1440
1450
1460
1470
1480
1490
1500
1510
1520
1530
1540
1550
1560
1570
1580
1590
1600
1610
1620
1630
1640
1650
1660
1670
1680
1690
1700
1710
1720
1730
1740
1750
1760
1770
1780
1790
1800
1810
1820
1830
1840
1850
1860
1870
1880
1890

C

```

DC 70 N=1,49,2
UNIT = DFLOAT(N)*VW/2.D0
X = SLOPE*UNIT+VMPTR
Y = 0.0D0
Z = (VDIS+UNIT)*(-1.D0)
VCORD = VRC*(1.D0-TAPR*UNIT/VB)
VTHIK = VRT*(1.D0-THICKR*UNIT/VB)
VOLV = VW*VCORD*VTHIK/(12**3)
DELMV = VOLV*DENV
WEIGHV = DELMV+WEIGHV
PRODV = DELMV*X+PRODV
PRODMZ = DELMV*Z+PRODMZ
A4 = DELMV*(Y**2+Z**2)+A4
B4 = DELMV*(X**2+Z**2)+B4
C4 = DELMV*(X**2+Y**2)+C4
E4 = DELMV*X*Z+E4
70 CONTINUE

```

C C C

VB1 = VB+VDIS

C C C

COMPUTATION OF C.G. AND MOMENTS OF INERTIA DUE TO ANY LUMPED MASSES

```

READ (5,140) N
WEIGHM = 0.0D0
PRODM = 0.0D0
PRODMV = 0.0D0
PRODMZ = 0.0D0
A5 = 0.0D0
B5 = 0.0D0
C5 = 0.0D0
E5 = 0.0D0

```

C C C

```

DO 80 I=1,N
READ (5,160) W,X,Y,Z
WM(I) = W
XM(I) = X
YM(I) = Y
ZM(I) = Z
WEIGHM = W+WEIGHM
PKODM = W*X+PRODM
PRODMV = W*Y+PRODMV
PRODMZ = W*Z+PRODMZ
A5 = W*(Y**2+Z**2)+A5

```



```

WRITE (6,450) WB
WRITE (6,460) WA
WRITE (6,470) SWLEW
WRITE (6,480) GAMAW
WRITE (6,490) CLAW
WRITE (6,500) CRT1
WRITE (6,430) CTT
WRITE (6,440) TB
WRITE (6,510) ST
WRITE (6,520) SWLET
WRITE (6,470) GAMAT
WRITE (6,480) CLAT
WRITE (6,490) VRC1
WRITE (6,530) VTC
WRITE (6,430) VB1
WRITE (6,540) VS
WRITE (6,520) SWLEV
WRITE (6,470) CLAV
GO TO 110
100 READ (5,120) STAR
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C COMPUTATION OF RELEVANT LONGITUDINAL STABILITY COEFFICIENTS C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
READ (5,150) H,CLTRM
READ (5,130) ST
READ (5,160) TL,TH,CLAT,CMOWB
READ (5,160) WA,WB,HNWB,SWLEW
READ (5,160) CRI,CT,GAMAW,ALFAZL
READ (5,160) CLAW,CLAWS,CLAM,CLAWB
RADIAN = 57.29577DO
SWLE = SWLEW/RADIAN
GAMA = GAMAW/RADIAN
AR = WB**2/WA
TAP = CT/CRI
TSWQC = DTAN(SWLE)-(4.DO/AR)*(.25DO*((1.DO-TAP)/(1.DO+TAP)))
SWQC = DATAN(TSWQC)
AK = 1.DO/AR-1.DO/(1.DO+AR**1.7)
AL = (10.DO-3.DO*TAP)/7.DO
POWER = 1.DO/3.DO
AH = (1.DO-TH/WB)/(2.DO*TL/WB)**POWER
CSWQC = DCOS(SWQC)
DEDA = 4.44DO*(AK*AL*AH*DSQRT(CSWQC))**1.19

```

2860
2870
2880
2890
2900
2910
2920
2930
2940
2950
2960
2970
2980
2990
3000
3010
3020
3030
3040
3050
3060
3070
3080
3090
3100
3110
3120
3130
3140
3150
3160
3170
3180
3190
3200
3210
3220
3230
3240
3250
3260
3270
3280
3290
3300
3310
3320
3330


```

OMDEDA = 1.D0-DEDA
CBARG = ((2.D0*CR1)/3.D0)*((1.D0+TAP+TAP**2)/(1.D0+TAP)
VH = (TL*ST)/(WA*CBARG)
BRCKT1 = CLAT*OMDEDA/CLAWB
BRCKT2 = VH*BRCKT1
BRCKT3 = H*ST*BRCKT1/WA
UPI = HNW*BRCKT2+BRCKT3
DWN1 = 1.D0+BRCKT3/H
HN = UPI/DWN1
SFSLN = HN-H
EPSLN0 = -DEDA*ALFAZL/RADIAN
EPSLN1 = EPSLN0*RADIAN
CMA = -CLAM*SFSLN
CMQ = -2.D0*CLAT*.9D0*VH*TL/CBARG
CMAD = CMQ*DEDA
CMAT = -VH*CLAT
AR = WB**2/WA
TSUB1 = -ALFAZL/RADIAN
PAREN1 = CLAT*TL*ST*(EPSLN0+TSUB1)/(WA*CBARG)
PAREN2 = CLAWB*CLAT*ST*(EPSLN0+TSUB1)/(CLAM*WA)
PAREN3 = H-HNW
PAREN4 = (TL*ST*CLAT*(1.D0-DEDA))/(CLAWB*WA*CBARG)
PAREN5 = PAREN3-PAREN4
CMQA = CMQ*PAREN1+PAREN2*PAREN5
CONST = TL/CBARG
CLTRIM = CLTRM*CONST
ALFAM = (CLTRIM-CMQA)/(CLAM*CONST+CMA)
ALFAT = (CLTRIM-CLAM*ALFAM)/(CLAT*ST/WA)
ALFAW1 = ALFAM*RADIAN
ALFAT1 = ALFAT*RADIAN
ALFAWB = ALFAM*(CLAT*ST*(EPSLN0+TSUB1))/(CLAM*WA)
CLWB = CLAWB*ALFAWB
GO TO 90

```

110

```

WRITE (6,550)
WRITE (6,560)
WRITE (6,570)
WRITE (6,580)
WRITE (6,590)
WRITE (6,600)
WRITE (6,610)
WRITE (6,620)
WRITE (6,630)

```

```

CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C COMPUTATION OF THE ELEMENTS OF THE STATE VARIABLE MATRICES
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC

```



```

C
AIX = (8.0D0*A)/(2.378D-3*WA*32.2D0*WB**3)*(12**3)
BIY = (8.0D0*B)/(2.378D-3*WA*32.2D0*CBAR*3)*(12**3)
CIZ = (8.0D0*C)/(2.378D-3*WA*32.2D0*WB**3)*(12**3)
EIXZ = (8.0D0*E)/(2.378D-3*WA*32.2D0*WB**3)*(12**3)
A11 = (CMQ+CMAD)/BIY
A21 = 1.0D0
A22 = 0.0D0
BM1 = CMAT/BIY
BM2 = 0.0D0
WRITE (6,640)
WRITE (6,650) A11,A12
WRITE (6,660) A21,A22
WRITE (6,670)
WRITE (6,680) BM1
WRITE (6,680) BM2
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C COMPUTATION OF RELEVANT LATERAL STABILITY DERIVATIVES C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
WRITE (6,550)
WRITE (6,690)
WRITE (6,700)
WRITE (6,710)
WRITE (6,720)
WRITE (6,730)
WRITE (6,740)
WRITE (6,750)
WRITE (6,760)
WRITE (6,770)
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C COMPUTATION OF CL(BETA) AND CN(BETA) C
C
READ (5,180) CLBCLS,SKM,FK,CLBCLA,CLBG,GKM
READ (5,150) H,ZW
RADIAN = 57.29577D0
CLBWB1 = CLBCLS*SKM*FK+CLBCLA
CLBWB2 = CLWB*CLBWB1
DELCLB = -5.0D-4*AR*(DISTW/WB)**2
CLBWB3 = CLBG*GKM+DELCLB
CLBWB4 = GAMAW*CLBWB3
DCLBZW = (1.2D0*DSQRT(AR)*ZW*2.D0*DISTW)/(RADIAN*WB**2)
CLBWB = (CLBWB2+CLBWB4+DCLBZW)*RADIAN
READ (5,180) HCBCLS,HCKM,HFK,HCBCLA,HCLBG,HGKM
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```



```

4300 READ (5,130) ZWH
4310 CLTAIL = CLAT*ALFAWB*(1.D0-DEDA)-CLAT*(EPSLNO+TSUBI)
4320 CLBH1 = HCBCLS*HSKM*HFK+HCBCLA
4330 CLBH2 = CLTAIL*CLBH1
4340 ART = TB**2/ST
4350 DHCLBZ = (1.2D0*DSQRT(ART)*ZWH*2.D0*DIST)/(RADIAN*TB**2)
4360 DHCLBG = -5.D-4*ART*(DIST/TB)**2
4370 CLBH3 = HCLBG*HGKM+DHCLBG
4380 CLBH4 = GAMAT*CLBH3*RADIAN
4390 VH1 = (ST*TB)/(WA*WB)
4400 CLBH5 = RADIAN*(CLBH2+CLBH4+DHCLBZ)
4410 CLBH = VH1*CLBH5
4420 READ (5,170) CONSTV,CLAV,VS
4430 READ (5,150) ZV,VL
4440 CSWQC = DCOS(SWQC)
4450 DSDB = .724D0+3.06D0*(VS/(WA*(1.D0+CSWQC)))+.4D0*ZWH/DISTW+9.D-3*A
4460
4470 1R CYBV = -CONSTV*CLAV*DSDB*VS/WA
4480 CLBV = CYBV*(ZV*DCOS(ALFAWB)-VL*DSIN(ALFAWB))/WB
4490 CLB = CLBW+CLBH+CLBV
4500 VLC = VL*DCOS(ALFAWB)+ZV*DSIN(ALFAWB)
4510 ZVC = ZV*DCOS(ALFAWB)-VL*DSIN(ALFAWB)
4520 READ (5,160) XKN,XKRL,SBS,FL
4530 CNB8 = -RADIAN*XKN*XKRL*SBS*FL/(WA*WB)
4540 CNBV = -CYBV*VLC/WB
4550 CNB = CNB8+CNBV
4560 WRITE (6,780) CLB,CNB
4570
4580 C COMPUTATION OF CL(P) AND CN(P)
4590
4600 READ (5,160) XBCLPK,CLAWS,HBCLPK,CLATS
4610 CCNW = CLAWS/(2.D0*3.14159D0)
4620 CCNT = CLATS/(2.D0*3.14159D0)
4630 CLPW = XBCLPK*CCNW
4640 CLPV = 5D0*(HBCLPK*CONT)*(ST/WA)*(TB/WB)**2
4650 CLP = CLPW+CLPV+CLPV
4660 READ (5,130) XBARI
4670 TSWQC = DTAN(SWQC)
4680 B7 = 1.D0
4690 TANOAT = DTAN(ALFAWB)
4700 AB7 = A*B7
4710 CNPW1 = -CLPW*TANOAT
4720 CNPW2 = -CLP*TANOAT
4730 CNPW31 = (AR+4.D0*CSWQC)/(AB7+4.D0*CSWQC)
4740 CNPW3T = AB7+.5D0*(AB7+CSWQC)*(TSWQC)**2
4750 CNPW3B = AR+.5D0*(AR+CSWQC)*(TSWQC)**2
4760 CNPW3 = (CNPW3T/CNPW3B)*CNPW31
4770

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CNPW41 = 6.D0*(AR+CSWQC)
CNPW42 = (XBAR1*TSWQC)/(CBARG*AR)+(TSWQC)**2/12.D0
CNPW43 = -1.D0*(AR+CNPW41*CNPW42)
CNPW48 = 6.D0*(AR+4.D0*CSWQC)
CNPW4 = CNPW41/CNPW48
CNPW5 = CNPW3*CNPW4
CNPW6 = CNPW5*CLWB
CNPV = CNPW1-(CNPW2-CNPW6)
CNPV = -2.D0*VLC*(ZVC-ZV)*CYBV/WB**2
CNP = CNPW+CNPV
WRITE (6,790) CLP,CNP

C
C
C      COMPUTATION OF CL(R) AND CN(R)
C
READ (5,130) XCLRCL
CTOP1 = AR*(1.D0-B7**2)/(2.D0*B7*(AB7+2.D0*CSWQC))
CTOP2 = ((AB7+2.D0*CSWQC)*TSWQC**2)/(8.D0*(AB7+4.D0*CSWQC))
CTOP = 1.D0+CTOP1+CTOP2
CBOT = 1.D0+(AR+2.D0*CSWQC*TSWQC**2)/(8.D0*(AR+4.D0*CSWQC))
CLRW1 = (CTOP/CBOT)*XCLRCL*CLWB
DCLRG = (3.14159D0*AR*DSIN(SWQC))/(12.D0*(AR+4.D0*CSWQC))
CLRW = CLRW1+GAMA*DCLRG
CLRV = -2.D0*VLC*ZVC*CYBV/WB**2
CLR = CLRW+CLRV
READ (5,170) XCNRCL,XCNRCD,CDO
CNRW = XCNRCL*CLWB**2+XCNRCD*CDO
CNRV = 2.D0*(VLC**2)*CYBV/WB**2
CNR = CNRW+CNRV
WRITE (6,800) CLR,CNR

C
C
C      COMPUTATION OF CL(DELTA-AILERON) AND CN(DELTA-AILERON)
C
READ (5,160) BCPLDK,CLD1D,CLDT,CK
CONSTK = CLAWS/(2.D0D*3.14159D0)
CPLD = CONSTK*BCPLDK
CLD1 = CLD1D*CLDT
ALFAD = CLD1/CLAWS
CLDEL = ALFAD*CPLD
CLDA = 2.D0*CLDEL
CNDA = CK*CLWB*CLDA
WRITE (6,810) CLDA,CNDA

C
C
C      COMPUTATION OF CL(DELTA-FIN) AND CN(DELTA-FIN)
C
CLDF = (CLAV*VS*ZV)/(WA*WB)
CNDF = -(CLAV*VS*VL)/(WA*WB)
WRITE (6,820) CLDF,CNDF
C

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CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C  COMPUTATION OF ELEMENTS FOR STATE VARIABLE MATRICES (LATERAL) C
C  CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C

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DNCMTR = AIX*CIIZ-EIXZ**2
C11 = (CIIZ*CLP+EIXZ*CNP)/DNOMTR
C12 = (CIIZ*CLR+EIXZ*CNR)/DNOMTR
C13 = (CIIZ*CLB+EIXZ*CNB)/DNOMTR
C14 = 0.000
C21 = (AIX*CNP+EIXZ*CLP)/DNOMTR
C22 = (AIX*CNR+EIXZ*CLR)/DNOMTR
C23 = (AIX*CNB+EIXZ*CLB)/DNOMTR
C24 = 0.000
C31 = 0.000
C32 = -1.000
C33 = 0.000
C34 = 0.000
C41 = 0.000
C42 = 0.000
C43 = 0.000
C44 = 0.000
D11 = (CIIZ*CLDA+EIXZ*CNDA)/DNOMTR
D12 = (CIIZ*CLDF+EIXZ*CNDF)/DNOMTR
D21 = (AIX*CNDA+EIXZ*CLDA)/DNOMTR
D22 = (AIX*CNDF+EIXZ*CLDF)/DNOMTR
D31 = 0.000
D32 = 0.000
D41 = 0.000
D42 = 0.000
WRITE(6,640)
WRITE(6,830)
WRITE(6,830)
WRITE(6,830)
WRITE(6,830)
WRITE(6,840)
WRITE(6,850)
WRITE(6,850)
WRITE(6,850)
WRITE(6,850)
WRITE(6,550)

```

```

C11,C12,C13,C14
C21,C22,C23,C24
C31,C32,C33,C34
C41,C42,C43,C44
D11,D12
D21,D22
D31,D32
D41,D42

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120 FORMAT (5A4)
130 FORMAT (F10.7)
140 FORMAT (I2)

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1 1//)
620 FORMAT (43X, 'CMQ = ', 5X, 1PD14.7, 4X, 'CMA = ', 6X, 1PD14.7, //)
630 FORMAT (43X, 'CMA(DOT) = ', 1PD14.7, 4X, 'CMA(TAIL) = ', 1PD14.7, //)
640 FORMAT (43X, 'ELEMENTS OF STATE VARIABLE MATRIX A ARE:', //)
650 FORMAT (54X, 1PD14.7, 5X, 1PD14.7, //)
660 FORMAT (54X, 1PD14.7, 5X, 1PD14.7, //)
670 FCORMAT (43X, 'ELEMENTS OF THE B VECTOR ARE:', //)
680 FCORMAT (64X, 1PD14.7, //)
690 FCORMAT (56X, 'LATERAL STABILITY DERIVATIVES:')
700 FCORMAT (56X, 10( '---'), //)
710 FCORMAT (48X, 'CL- = VARIATION OF ROLLING MOMENT COEFFICIENT', //)
720 FCORMAT (54X, 'WITH -', //)
730 FCORMAT (48X, 'CN- = VARIATION OF YAWING MOMENT COEFFICIENT', //)
740 FCORMAT (54X, 'WITH -', //)
750 FCORMAT (48X, 'B = SIDESLIP ANGLE (BETA)', //)
760 FCORMAT (48X, 'P = ROLL RATE', 17X, 'R = YAW RATE', //)
770 FCORMAT (48X, 'DA = AILERON DEFLECTION', 7X, 'DF = RUDDER DEFLECTION',
1 //)
780 FCORMAT (45X, 'CLB = ', 1PD14.7, 10X, 'CNB = ', 1PD14.7, //)
790 FCORMAT (45X, 'CLP = ', 1PD14.7, 10X, 'CNP = ', 1PD14.7, //)
800 FCORMAT (45X, 'CLR = ', 1PD14.7, 10X, 'CNR = ', 1PD14.7, //)
810 FCORMAT (45X, 'CLDA = ', 1PD14.7, 10X, 'CNDA = ', 1PD14.7, //)
820 FCORMAT (45X, 'CLDF = ', 1PD14.7, 10X, 'CNDF = ', 1PD14.7, //)
830 FCORMAT (42X, 4( 1PD14.7, 4X), //)
840 FCORMAT (1//, 43X, 'ELEMENTS OF THE B MATRIX ARE:', //)
850 FCORMAT (54X, 2( 1PD14.7, 5X), //)
END

```


LIST OF REFERENCES

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3. Roskam, J., Methods for Estimating Stability and Control Derivatives of Conventional Subsonic Airplanes, published by the author, 519 Boulder, Lawrence, Kansas, 1971.

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